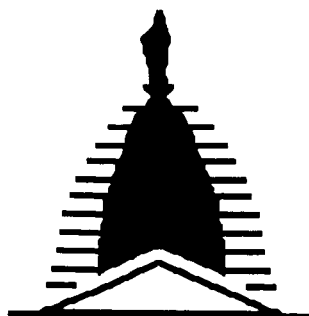


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**UNIVERSITY of
NOTRE DAME**

(NASA-CR-186215) THE AIR RHINO: AN RPV
DESIGNED TO INVESTIGATE FORCES AND MOMENTS
ON A LIFTING SURFACE Final Design Proposal
(Notre Dame Univ.) 120 p

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**NASA/USRA UNIVERSITY
ADVANCED DESIGN PROGRAM
1988-1989**

Final Design Proposal

THE AIR RHINO

**An RPV Designed to Investigate
Forces and Moments on a Lifting Surface**

**Department of Aerospace and Mechanical Engineering
University of Notre Dame
Notre Dame, IN 46556**

Air Rhino

--- design proposal for a remotely piloted airfoil test platform ---



Submitted by Group E in response to Gold Mission Request For Proposals

Group E members:

Scott Barton
Paul Bielski
Debbie Dooley
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Joe Haudrich
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Mark McLaughlin
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Tony Villanueva

The Design Team . . .

Air Rhino
Design Proposal for
A Remotely Piloted Airfoil Test Platform

submitted May 2, 1989 in response to "Gold Mission - Request for Proposals"
University of Notre Dame Aerospace and Mechanical Engineering

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This document set in Helvetica and New Century Schoolbook
on random Macintoshes throughout the world

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Executive Summary

The Air Rhino is a remotely piloted "airfoil test platform" airplane. It is designed to measure the component forces on easily interchangeable test airfoils for low Reynolds' numbers and varying angles of attack. This type of data is difficult or impossible to collect in conventional wind tunnels.

A three-view drawing and specifications summary follow this executive summary.

The data will be gathered in a steady flight environment. The flight plan calls for ascent to cruise altitude and, once there, flying many straight, level, unaccelerated test runs where data will be taken. The flight is kept straight and true by an autopilot system. While the plane is circling back for another test run, the pilot may make adjustments in test airfoil angle of attack or autopilot programmed flight velocity.

Measurements of the lift, drag and moment on the vertically mounted test airfoil will be made using a force balance. Other measurements to be taken include the static pressure, the dynamic pressure, the temperature, the plane angle of attack, and the test airfoil angle of attack. This data is sent back to a ground-based receiver, where a custom-built circuit board converts the data into a format easily understood and manipulated by microcomputers.

The propulsion system consists of a pusher propeller mounted behind the test airfoil and the fuselage to avoid flow interference on the test section. The prop, a three-bladed Clark-Y, is powered by a Quadra Q-82 reciprocating gas engine capable of producing 8 Hp @ 8000 rpm. The three-bladed propeller was chosen for efficiency and noise reduction, and the engine was chosen for its high power output and simplicity of maintenance. Endurance, range, rate of climb and rate of descent are all excellent for the Air Rhino, because its propulsion system is geared towards the top velocity of 200 ft/s, so the engine is overpowered for the middle speed ranges.

Air Rhino obtains its lift force from a 9.83 ft span, 1.5 ft mean chord wing with spar-and-rib construction. The wing skin will be a mylar-based derivative like Monokote. Spars and ribs may be built out of either thin aluminum or thicker and less expensive woods like spruce and balsa.

Special care must be taken to insure the impact loads on landing do not substantially destroy the fragile instrumentation.

The tail and horizontal stabilizer are located at the end of long twin booms mounted aft of the propeller. The booms must be long for two reasons: a) to keep the surfaces out of the worst of the draft effects, and b) a longer boom assembly yields more moment to compensate for large forces on the test section. These surfaces are oversized to help compensate as well.

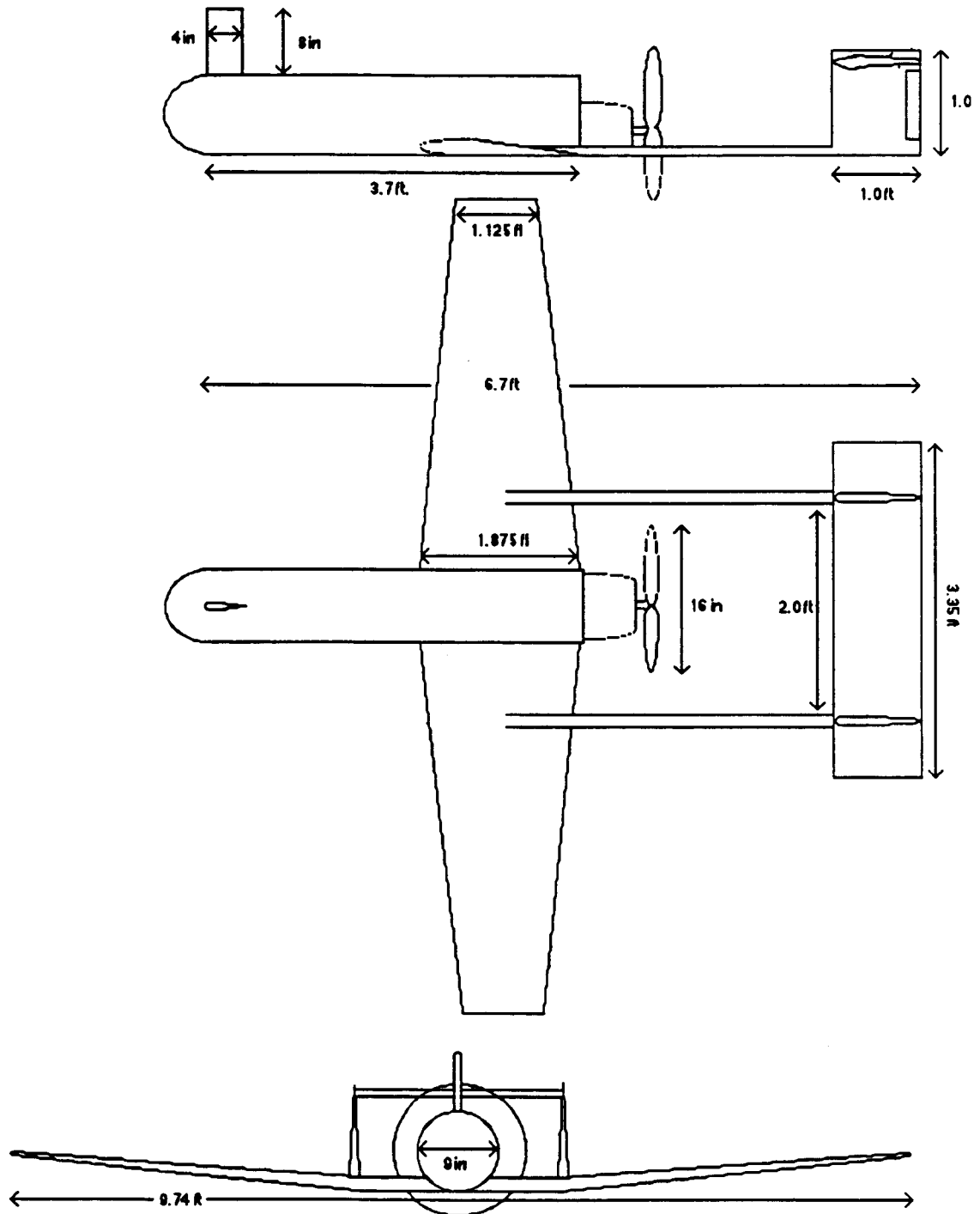
Three movable control surfaces are used to control the aircraft: ailerons to change roll, a rudder to change yaw, and a stabilator to change pitch. These surfaces are actuated by micro-servos, as are the flaps, the landing gear and the throttle.

Some areas that will require future study are the accurate determination of static margin, compensation for both the moment and the force of the test airfoil, the reduction of landing distance, and the details of fuselage surface flow.

After fourteen weeks of hard work and discussion, we feel we have come up with a design that will fulfill the requirements of the Gold Mission reasonably well. While there are acknowledged problems with the design to be solved, we feel that the Air Rhino concept is worthy of continued support and eventual production.

- Design Group E
designers of Air Rhino

3-View Drawing of Air Rhino



Specifications Summary

specification	range	symbol
Endurance	30 min @ 200 ft/sec 101 min @ 55 ft/sec	E
Engine type	Quadra Q-82	P
" " power	8 Hp @ 8000 RPM	
Fuselage diameter	9 in	Std
" " length	3.7 ft	
Landing distance	341 ft	Std
Max load factor	2.5	n
Weight fully loaded	35.3 lb	W
Propeller type	16 in. 3-blade Clark Y	e _{prop}
" " efficiency	0.86	
Range	100 miles @ 80 ft/sec	R
Reynolds number range	136000 -- 660000	Re
Speed range	41 -- 200 ft/sec	V _{stall} -- V _{max}
Takeoff distance	210 ft	S _{to}
Test airfoil chord	4 in	c _{test}
" " maximum angle of attack	-20° -- +30°	a
" " maximum lift	35 lb	L _{test}
" " maximum drag	7 lb	D _{test}
" " aspect ratio	2	
Wing surface area	15 sq ft	S
" " root chord	1.875 ft	K
" " taper ratio	0.6	
" " mean chord	1.5 ft	
" " span	9.74 ft	b
" " aspect ratio	6.33	AR
" " max lift coefficient	0.72	CL _{max}
" " dihedral	5°	
" " angle of incidence	1°	
Flap size	60% span, 20% chord, max deflection 40°	
" " lift coefficient increment	0.5	
" " drag coefficient increment	0.06	
Vertical tail area	2 tails @ 1 sq ft	S _v
Horizontal tail area	3.35 sq ft	S _h

Part I. Preliminary Concept

I. Preliminary Concept Design

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Review of Design Requirements

The following document was provided as design specifications for an remotely piloted vehicle (RPV) to be created by Design Group E:

AERODYNAMIC DATA ACQUISITION USING REMOTELY PILOTED VEHICLES MISSION - COMPONENT FORCES

OPPORTUNITY

The wind tunnel has served as the primary source of aerodynamic data for flight vehicle configurations. The wind tunnel is used to test subscale models of flight vehicles or to collect basic aerodynamic data. Within the wind tunnel certain flow conditions can be accurately controlled such as model position and flow speed but other influences such as wall interference and free stream turbulence are more difficult, if not impossible, to control. Wind tunnel testing can also be limited by the ability to achieve dynamic similarity between the test and actual flight conditions. Remotely piloted vehicles have been used for testing technology demonstrators but their use in collecting in-flight aerodynamic data has not yet been fully exploited. The use of a RPV for the collection of aerodynamic data at low Reynolds numbers will be the goal of this design effort.

OBJECTIVES

1. Develop a proposal for an aircraft and associated flight control and data acquisition system which must be able to:
 - a. Be used as an airborne aerodynamic data acquisition system for collecting component flight load data for both the primary lifting surfaces and the horizontal and vertical stabilizers. The goal will be to provide the means for evaluating influence of different airfoil sections on these lifting surfaces.
 - b. The flight vehicle must be able to operate at angles of attack from -20° to +40° over a Reynolds number range (based on wing root chord) of 40×10^4 to 1×10^6 .
 - c. Considerations should be given for both rectangular and tapered wing planforms.
 - d. The instrumentation system must be capable of collecting all necessary associated data such as airspeed, angle of attack, control surface position, etc., with sufficient accuracy in order to provide useful aerodynamic information.
2. Take full advantage of the latest technologies associated with lightweight, low cost radio controlled aircraft and propulsion systems. Since this system may be expected to operate in a wide variety of climates and test conditions the safety of the system will be of critical importance. All possible considerations must be taken to avoid damage or injury in case of system malfunction.

SYSTEM REQUIREMENTS AND CONSTRAINTS

The system design shall satisfy the following.

- a. All basic operation will be line-of-sight although automatic control or other systems can be considered.
 - b. Takeoff and landing must be accomplished in a circular area with no greater than a 150 ft radius. (50 ft object clearance). Any special landing or takeoff equipment must be considered as part of the system. For repeated flights system turnaround must be able to be accomplished in 15 min.
 - c. Only clear weather capabilities need be considered. In order to evaluate the influences of wind and gusts, wind speeds of up to 20 mph with a linear gradient from 10 fps at 0 feet (altitude) to 25 fps at 100 feet and constant speed of 25 fps above that altitude. Gusts of 10 fps should be considered.
 - d. All airborne instrumentation and associated flight control system must be included in the design.
 - e. Ground handling and system operation must be able to be accomplished by two people.
 - f. The complete system should be portable in a conventional pickup truck.
 - g. Noise nuisance must be a consideration both for the operator and the region in which the aircraft operates.
-

According to this document, the design group was to set out to measure component forces on surfaces in flight. These forces were interpreted as being the lift, drag and moment forces acting on these surfaces. Design Group E chose to limit test measurements to one aerodynamic surface -- a wing, a tail, or a horizontal stabilizer -- during any one test flight. Rigging a plane to measure component forces on all three of these surfaces at once would be expensive and extremely difficult to design.

The requirement that different airfoils and planforms be evaluated was one of the main physical considerations in the design of the craft and this condition was included in every considered concept.

After preliminary design attempts of an aircraft to attempt to meet these parameters, it was decided to limit some of the parameters to more realistic or easier-to-attain values. Simple calculations show that the required aircraft speed, given a test airfoil with smallest practical chord length of 4 inches, is $V = \frac{v^* Re}{L} = 300 \text{ ft/sec}$ for $Re=1,000,000$ and $V=12 \text{ ft/sec}$ for $Re=40,000$. Allowing different test airfoil chords was ruled out because of difficulties in control sizing and attachment to the plane. The Reynolds

number range was therefore changed to 1.6×10^5 -- 6.6×10^5 , allowing design of a plane with the more reasonable speed range of 50 ft/sec to 200 ft/sec. This seemed the best compromise on the tradeoff between stall speed and top speed.

The acceptable range of angle of attack was reduced from -20° through $+40^\circ$ to -20° through $+30^\circ$. This reduction cut down the drag forces associated with test surface. This was also considered a more practical range as most airfoils rarely achieve $+40^\circ$ without stalling. Data above the stall angle would be of limited use, and above-stall phenomena such as the stall hysteresis effect could not be measured from an airborne platform anyway because of vibrations and other instabilities.

The takeoff and landing distance requirement was extended after calculations showed that a 375 ft landing distance would be required. Takeoff distances were only slightly smaller.

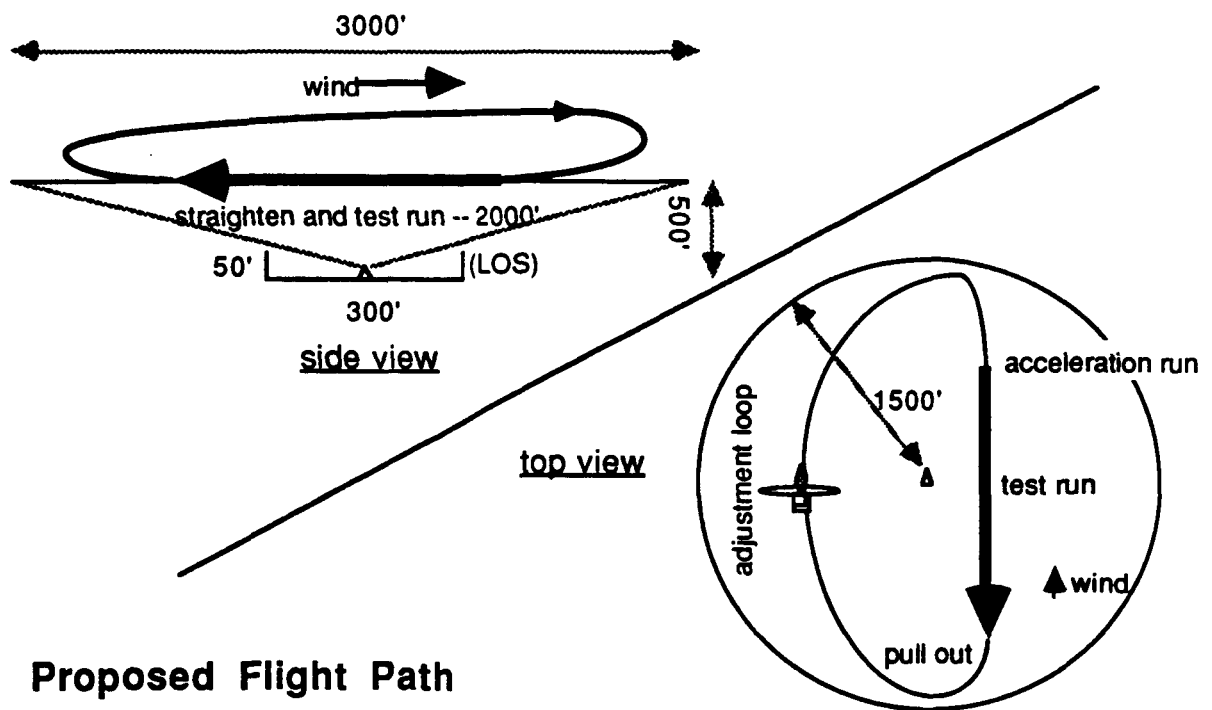
These three specifically mentioned modifications of the given requirements are the only special considerations made in this design. All the other requirements have been met.

Mission Definition and Flight Path

The mission of the Air Rhino is essentially to collect accurate force component data for aerodynamic surfaces over a low Reynolds number range. In essence, the Air Rhino must duplicate the wind tunnel environment without the tunnel. It must provide a steady platform for taking consistent component force data, and fairly consistent motion of that platform to create a constant flow about the test surface.

This objective can be reached in one of two ways. The plane may shut off its propulsion system and attempt to glide in steady unaccelerated flight long enough to take readings. Alternatively, the plane may use some sort of feedback system to maintain a precise heading and speed, and attempt to ensure that disturbances from the propulsion system do not affect the data measurement.

Because of the high speeds required to reach the upper end of the desired Reynolds number range, the idle cruise method is not an option. Given a 4-inch chord length for the test section, the plane must achieve a speed of over 200 ft/sec and hold that speed for at least 5 seconds. So the aim of the mission is to set up and repeatedly execute a 'test run' of steady, level, unaccelerated flight for data measurement.



Given that aim, along with the object clearance and line-of-sight constraints on the system, the preceding flight plan was proposed.

The plane must first be able to leave the ground and clear the 50 foot object clearance with less than 300 feet horizontal travel. It must also land in the same space. This turns out to be quite a problem, especially for landing in such a precise area, and the Air Rhino as currently designed does not meet this requirement, landing within 370 feet.

Once off the ground, the plane must climb quickly to an altitude of about 500 feet. This altitude was chosen to give a fairly wide circle within which to fly; the line-of-sight restriction coupled with the 300 foot diameter, 50 foot wall allows a flight boundary of 3000 feet diameter. Higher altitudes will, of course, result in broader flight boundaries.

After the plane has reached cruise altitude, it will follow a "squashed oval" pattern (see above top view). As it moves around the curved side of the pattern, the Air Rhino will be slowed down to the optimal cruise speed of 55 ft/sec to allow time for adjustment of the test airfoil angle of attack and other parameters. The plane will also climb 50 feet or so to allow a downward approach to the test run. During the approach to the straight side, the plane will curve into a dive maneuver (see above side view) to accelerate up to the test run speed. There will then be a 2000 foot straight, level run into the wind, during which test data will be taken for up to 7 sec and sent back to the ground for analysis. As the plane reaches the end of the test run, it will start to turn back to the curved side of the oval, thus completing a loop.

The plane will be able to complete up to 40 of these runs in a 30 minute time span, and will have enough fuel left for landing and emergency maneuvers. It will then land, the force balance will be recalibrated, the plane refueled, and the systems checked for another flight.

Concept Selection Studies

During the preliminary concept stage, Design Group E as a group reviewed the design requirements. The design team discussed the relative importance of each requirement and came up with a prioritized listing of team design goals. This listing was formulated using a priority grid, similar to the one below, to compare one goal at a time to all the other goals.

EXAMPLE OF COMPARISON GRID RANKING

	weight	cost	Re range	landing	# occurrences
weight	-----	w	Re	landing	1
cost		-----	Re	landing	0
Re range			-----	Re	3
landing				-----	2

Each objective was compared in the grid to every other objective. Then the number of times each objective was given a priority was tallied, and the objectives were then ordered by the tally. For example, the objective 'Re range' would have been ranked first in the above grid, and the objective 'cost' would have been ranked last.

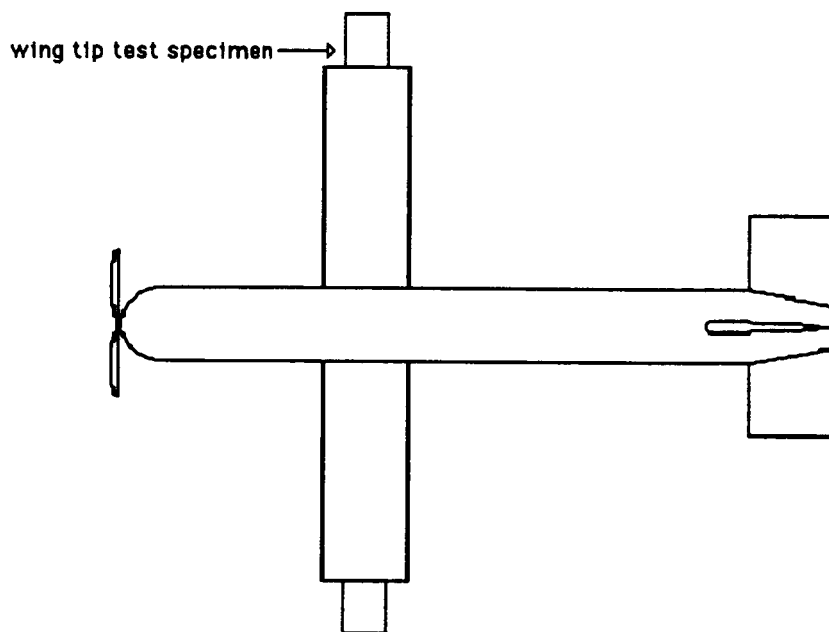
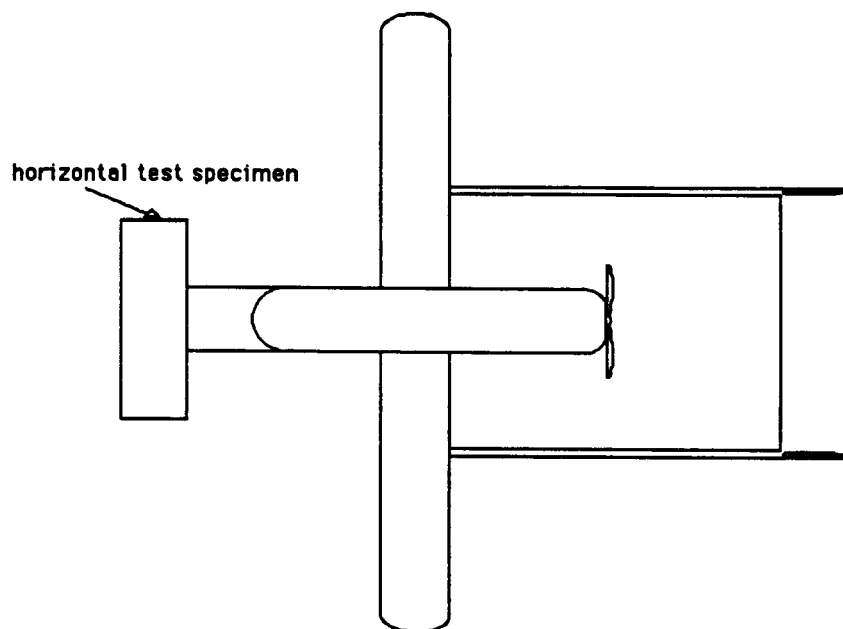
The results of Design Group E's comparison grid were as follows:

GROUP E DESIGN OBJECTIVES	
1.	measuring accuracy
2.	control/inherent stability/angle of attack
3.	landing/reusability
4.	Reynolds number range
5.	ground service/changing test airfoil
6.	safety/protect craft in case of malfunction
7.	endurance/rate of climb
8.	size (fit in pickup truck)
9.	low weight
10.	cost
11.	noise

These goals were agreed upon as the group design priorities, keeping in mind that many of these factors are interrelated and that any design will

eventually require some tradeoffs to be made.

Each member of the group submitted an individual design plan with sketches and a discussion of how the design goals could be achieved with the concept. While ten concepts were actually submitted, they seemed to fall into three broad categories. The eventual configuration of the Air Rhino was one of these three, and the other two are pictured below:



A fourth type of concept design was also submitted by one member. It proposed the use of the actual lifting surfaces of the aircraft to collect the component flight load data. A different "test" main wing would be installed on each flight. Each of these "test" wings would most likely have a different airfoil shape and characteristic lift curve slope.

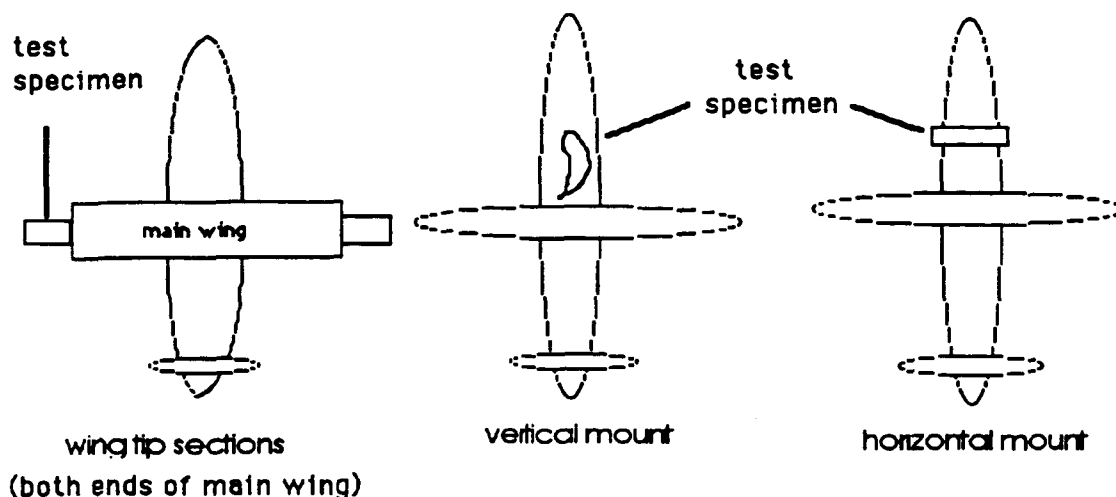
This concept proposed not meeting the full range of angle of attack (-20° to $+40^{\circ}$), reasoning that a record of performance of the airfoil beyond the stall angle would not be of practical use. Thus the RPV would be flown only in flight regions where the main wing would not stall, allowing for controlled flight.

This idea was rejected because of the following problems:

- Each testing cycle would require fabrication of a new full scale main wing that would have to be precisely fitted to the static aircraft configuration. The time and costs involved in continual fabrication of different full scale wings are undesirable.
- There is a question of how each main wing "test" airfoil would affect the ability of the RPV to become airborne. Different test wings will yield a variety of aerodynamic characteristics. The safety of the system could be jeopardized on each flight. This is not acceptable.
- In effect each "test" main wing would specify the testing flight regions as the stall conditions and thus the allowable flight environments vary from airfoil to airfoil.

The three remaining types of proposals featured an **extra** test airfoil attached to a complete flying craft. Measurements would be performed on the easily removable test airfoils. This was chosen as the best method for collecting the in-flight forces.

Three basic test airfoil locations were considered at length by Design Group E. Choice of the test airfoil location would have great effect on defining the rest of the RPV configuration. It would also influence the choice, placement and accuracy of devices to take force measurements, and the amount of flow interference experienced by the airfoil. Shown below are the proposed locations of the test airfoil:



In the wingtip mounting, the wingtip sections would connect directly into the wing structure with support bars. The vertical mount is similar in that it connects directly into the fuselage structure with a support bar. Horizontal mounting would be achieved by having one or two support posts secured to the fuselage structure on top of which the test specimen would be mounted.

The wingtip configuration looked easy to mount because the test airfoils could be connected directly to the wing structure. But the wingtips are susceptible to highly turbulent flow from the wing in the form of wingtip vortices. This flow would disrupt the flow environment for the test, which should be as consistent as possible. Loads caused by the test airfoils would add unnecessary stress to the wing structures. Furthermore, loads on the test airfoils would influence the aircraft greatly because of the long moment arm of the wing. The uncertainty of just what flows the test airfoil would experience and how test airfoil loads would affect the stability of the aircraft eliminated this configuration.

The horizontal mount was eliminated because the positioning of the test airfoil on posts would cause considerable problems. In this position, the airfoil could undergo considerable vibrations as a result of being on the end of a long post or posts. The posts could not be made shorter due to possible flow interference on the test section. The posts themselves would be subjected to large forces resulting from the resistance of the airfoil to the airflow; a post able to withstand these forces would add unnecessary weight. The long support posts combined with forces on the test airfoil

would also create longitudinal stability problems. Horizontal mounting was eliminated for these reasons.

In the end, a vertical mount for the test section was decided upon. In this configuration, the test specimen would be mounted on the top of the fuselage. To help create an undisturbed flow field near the test airfoil, a "pusher" propulsion system would be used. A large tail would help compensate for potential lateral stability problems without incurring lift penalties. The vertical mount also allows testing of large test airfoil angles of attack while maintaining level RPV flight, thus easing the burden on the propulsion system.

Placement of the test airfoil should be close to the center of gravity to minimize any moments it might exert on the craft. However, it is also desirable to mount the test airfoil very near the front of the craft to avoid upwash from the wing. A compromise of the two was reached.

Other design considerations collected from the many concepts are listed below :

- how to make the force measurements: accelerometer, pressure taps connected to a pressure transducer, control surface monitoring, or force balance;
- type and placement of propulsion system;
- range of Reynolds number to cover, and resultant speed range to cover;
- range of test airfoil angle of attack to cover;
- launch and retrieval systems;
- telemerty vs. onboard data storage

Each of these issues is discussed in the following paragraphs, followed by a drawing of the actual concept decided upon.

FORCE MEASUREMENT: One method proposed was the use of accelerometers, devices used to measure accelerations, to measure the change in lift and drag on an RPV due solely to movement of a test airfoil. Perhaps some method could be derived to calculate moment from these measures. The accelerometers could measure acceleration of the craft in either horizontal or vertical directions. But there would be no way to differentiate what portion of the forces on the aircraft had caused changes in acceleration. Any deflection of the test airfoil would cause the entire aircraft to accelerate in some direction and as a result the forces on all

parts of the plane change. Also, once a new equilibrium state is reached there is no acceleration to be measured. A change in the equilibrium forces could not be measured. Accelerometers were deemed unacceptable to acquire the flight load data because of these data separation problems.

Another suggested method of calculating the flight loads was to simply record the control surface deflections needed to maintain steady level flight when the test airfoil is at a certain angle of attack. The method was considered to present very complicated calculations. In addition, the aircraft position is changed when the control surfaces are deflected, and thus the entire network of body forces would change, requiring even more complex data reduction. This method was rejected.

A combined pressure ports and Pitot-static tube system was another means considered for force measurement. Calculations performed on the pressure readings attained by the pressure ports would give the lift force acting on the surface. A wake traverse by the Pitot tube will give the drag over the test airfoil. Pressure taps to take surface readings, each connected to tubing leading to a pressure transducer, are located across the entire chord of the test airfoil, with as many as 40 taps across the area feeding to 40 tubes. These would create problems in both volume occupied and time for measurement.

Force balances were presented as being a very good possibility for force measurement. Packages are known to exist that are very small and lightweight. The forces would be measured almost directly without any complicated calculations or mechanisms. Mission specific force balances would be relatively simple to construct and design. The device is relatively simple, calling for a few small strain gages on a flexible support spike hooked up to an electric meter device that would output voltage differences representative of the forces being experienced by the airfoil area. Data collected from the force balance could be read directly into a telemetry or onboard storage device.

PROPULSION SYSTEM: Because of the low stall speeds required to take measurements at low Reynolds numbers, most of the concepts proposed some sort of propeller-operated system. Possible engine choices included electric motors, "glowplug" motors, reciprocating gas engines, and any of the three with some sort of chemical rocket assist for high speeds. The rocket was vetoed as being too uncontrollable and unsteady,

and the lower-powered electric and glowplug motors were eliminated due to the high power required for high speeds. This left the reciprocating gas engine and propeller as the chosen propulsion system.

Because of the highly unsteady wake of the propeller, it was decided to mount it behind the test airfoil, creating a "pusher" plane.

MISSION REQUIREMENTS: The component forces on the test airfoil that must be measured are lift, drag, and moment.

From preliminary calculations it was decided that the entire Reynolds number range could not be covered. Instead we opted to try for a middle range of values. The high end of the range was considered to be not as important because there are other ways to gather data in that range. The low end of the range was likely to be limited by stall considerations.

The angle of attack range on the test airfoil was also limited, to lower magnitudes of the component forces and associated control problems. The high end of the range was eliminated because it was of limited practicality.

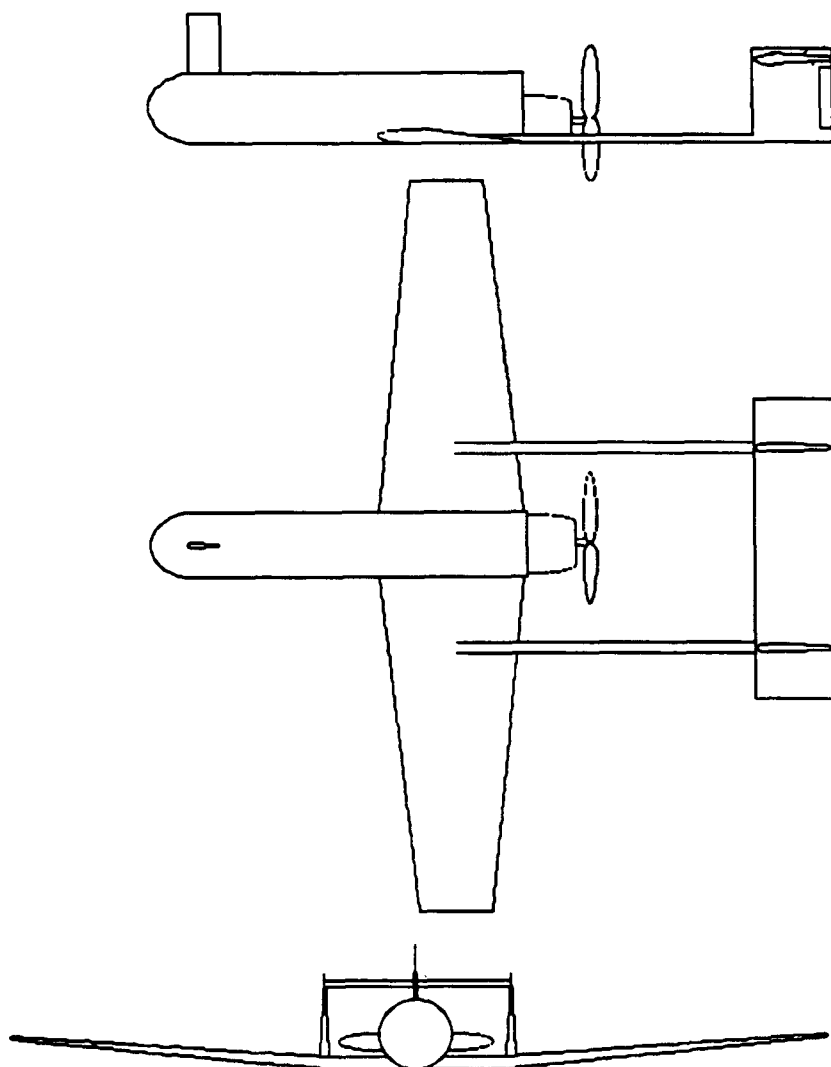
LAUNCH AND RETRIEVAL: Conventional landing gear, belly landing, parachute landing, net retrieval, balloon launch, and catapult launch were all considered. Landing was considered more critical because of the fragile electronics on board. Landing gear was eventually chosen for simplicity and low weight, and was used for conventional takeoff as well.

TELEMETRY AND DATA STORAGE: Onboard data storage was found to be more expensive and, more importantly, less reliable. The in-flight environment is demanding on electronics, with heat, unsteady power sources, and vibrations all causing problems. If the onboard device malfunctions, all data is lost. A telemetry system is somewhat more hardy, and will only stop data transmission without wiping previously acquired data. Telemetry was chosen.

OTHER CONSIDERATIONS: The group also clarified the details of the mission requirements for data acquisition. Certain measurements would be required for determination of flight conditions. Total and static pressure measurements along with temperature readings would be used to calculate the airspeed of the craft and the density of the air at that altitude. An inclination device would make measurements to determine the angle of attack of the craft, or a component of the craft. All this information, once radioed down to the ground for storage and analysis, would be used to determine the characteristics of the test surfaces.

For the data acquisition system, the lightest and most effective packages were sought out to minimize size and weight of the aircraft. Cost was considered to be a secondary concern for a craft of small size. Similar considerations were made for selection of materials, control systems and structural design.

The overall configuration presented below is presented as our solution to the defined mission requirements:



3-View Drawing of Preliminary Concept

Part II. Design

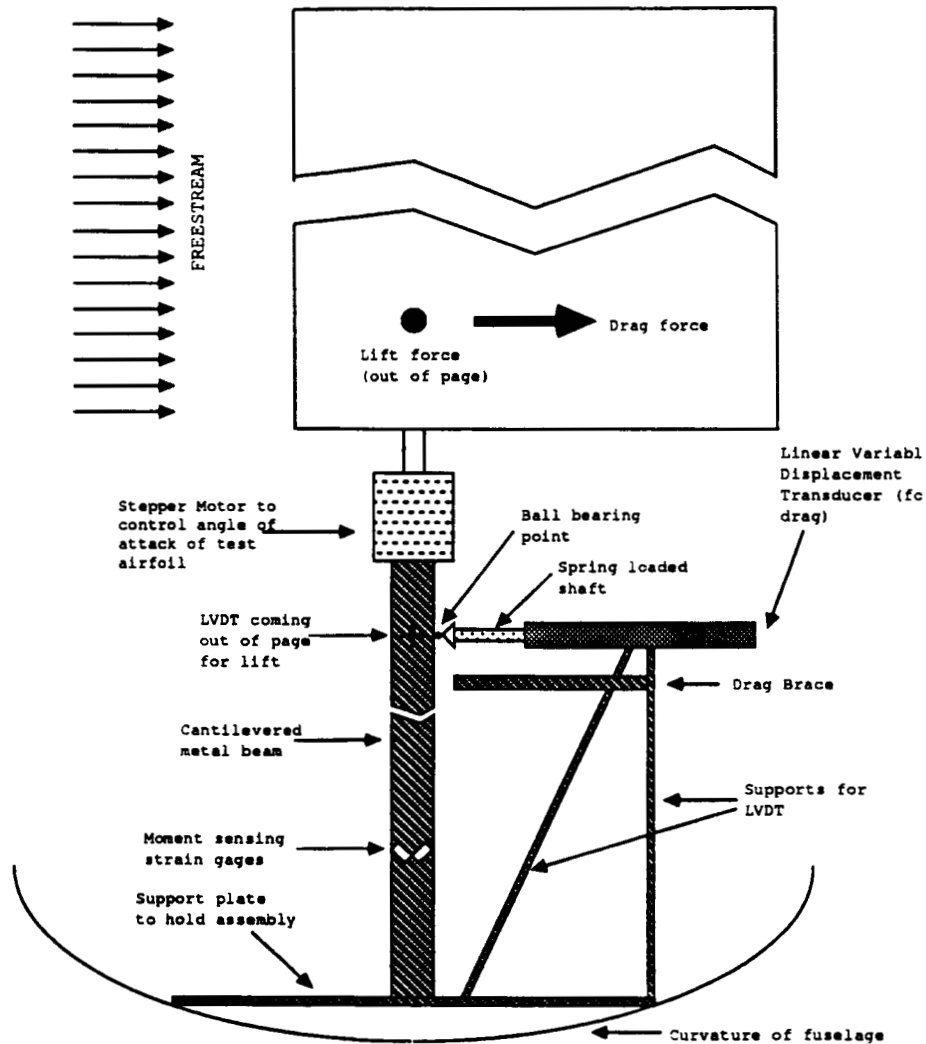
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Data Collection

The Force Balance

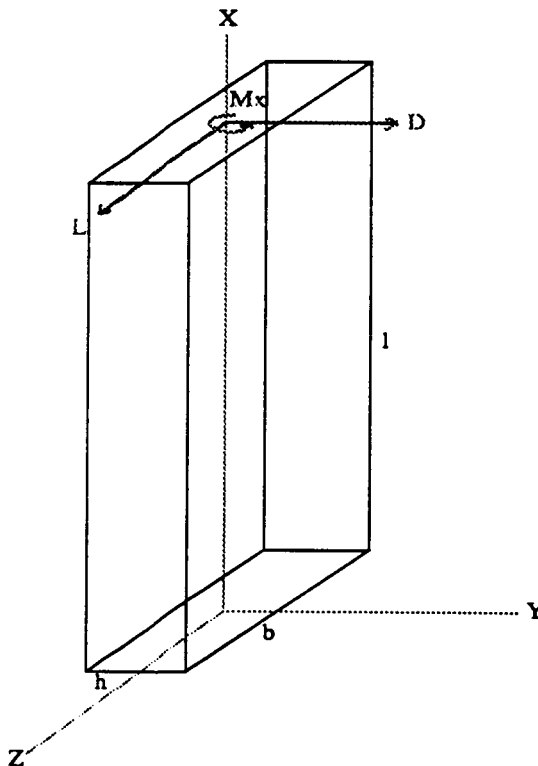
The concept chosen for the force measurement system is based on a very simple principle: the deflection of a beam or column of material is related to the amount of force applied to the column. The basic design is shown below.



The test airfoil will be attached to a stepper motor, which in turn will be attached to the solid, rectangular column of material. This column will then be fixed on a solid support plate of the same material. The stepper motor will be used to adjust the angle of attack of the test specimen in increments, such as a degree or half a degree. As the test airfoil undergoes

lift and drag forces during the flight, the column will deflect in two directions. These deflections will be measured by two linear variable displacement transducers (LVDT's), one in each of the two directions. Ball bearing tips on each of the LVDT's insure that the measurements in each direction are independent of movement in the other direction. Moment measurements will be made with two pairs of strain gages placed at 45 degree angles on two sides of the column. These four strain gages will be used in a Wheatstone Bridge configuration, making it insensitive to temperature while on the other hand making it very sensitive to changing strain in the column. The column can be calibrated very simply by applying known loads and moments, and then monitoring the output and curve fitting it. To prevent damage from loads above the design limit, support bars will prevent the column from deflecting too far.

Knowing the material properties, the position at which the force is applied, and the support conditions for the column, a relation can be determined for the deflection as a function of the force. The diagram below indicates the coordinate and naming system used for the derivation:



Shaft coordinates and dimensions

L , D , and M_x are the lift force, drag force, and moment about the x -axis respectively. The letters l , h , and b indicate the dimensions of the beam. For this case the equations for the deflection of the column are:

$$v_0 = \frac{2L}{Ehb^3}(3lx^2 - x^3)$$

$$w_0 = \frac{2D}{Ehb^3}(3lx^2 - x^3)$$

where v_0 and w_0 are deflections in the y and z directions respectively, E is the modulus of elasticity for the beam, and x is the distance along the x -axis at which the deflection is measured. The assumptions made in deriving these equations are that the beam is linear and homogeneous, there are no thermal loads, and the deflections are small. The beam is also cantilevered on a fixed support.

The dimensions and physical makeup of the column are the critical considerations for the balance, so a study was done to determine the best dimensions for the column, and the best placement along the column for the LVDT's. LVDT's of varying sensitivities can be purchased, and the support plate need only be big enough to support the column and fit in the plane. The figure of merit chosen was total weight for the column, the desired case being the lightest column that will satisfy the following constraints: the column must not yield under the maximum possible flight loads, and the dimensions must be such that the tips of the LVDT's do not "fall off" the edges of the column when it is deflected.

For the study, certain parameters were fixed. The length of the column was set at 9 inches due to the circular fuselage diameter of 9 inches. Actually, the top inch of the column would be replaced by the stepper motor, but for the purposes of the study it was assumed that the deflections of the column with the stepper motor would be almost the same as that of just a column of the same total height. The length of the column was chosen so that when it is mounted on its support plate, the top of the column would extend outside the fuselage for connection to the specimen. The airfoil specimen was assumed to be connected very close to the top of

the column, and the lift, drag, and moment experienced by the column were set at the maximum values of lift, drag, and moment for the airfoil specimen. This would be the case in which the column would deflect the most. Finally, the maximum deflections of the column in both the lift and drag directions were fixed at 0.1 inches. These deflections were made small so that the airflow around the specimen would change a very minimal amount when the column deflects.

Allowing for an extra tenth inch of travel positively and negatively for both the lift and the drag sensors, this then set the minimum dimensions of the column to $b = 0.4$ inches and $h = 0.3$ inches. The b dimension is larger because the lift sensor will actually move back and forth 0.2 inches due to positive and negative angle of attack situations. The drag sensor need only cover the 0.1 inch, since there is no case of "negative" drag.

There were three variables in the study: the material used for the column, the vertical position of the drag sensor (measured from the support base), and the vertical position of the lift sensor. Four materials were chosen that had a wide range of density and modulus of elasticity. They are 6061-T6 aluminum, 17-7PH steel, MIL-T-9047 titanium, and AZ80A-F magnesium. The vertical position of the drag sensor was varied between 4 and 7 inches (in the x direction). This range higher on the column was chosen to minimize the deflection of the top of the column, and therefore the specimen, because large movements of the specimen would affect the flow characteristics around it. The lightest possible beam would actually have the sensors placed very close to the base, but then the top would have very large deflections. The vertical position of the lift sensor was varied between 2.5 and 4.5 inches (in the x direction). This lower range had to be used because the curvature of the fuselage would interfere with the placement of the LVDT. The drag sensor does not have this problem because it is placed parallel to the axis of the fuselage.

From beam bending theory previously developed, the equations can be rearranged to provide the dimensions of the beam as functions of the maximum forces, and the placement of the sensors:

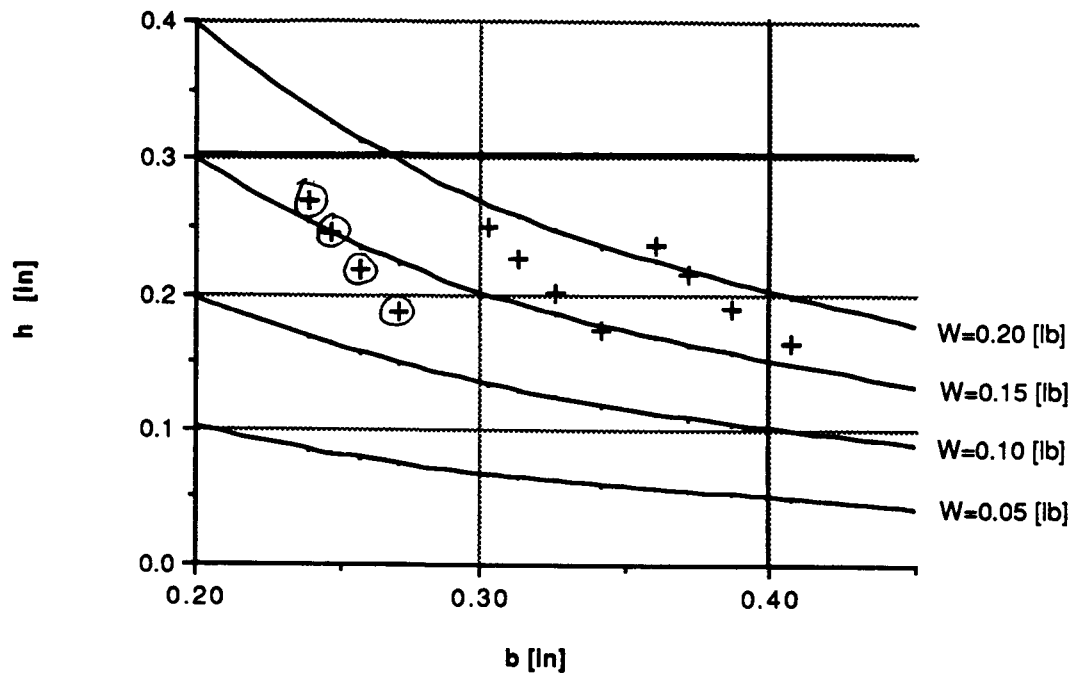
$$b = \left[\frac{4L^3 w_0 (3lx_L^2 - x_L^3)^3}{DE^2 v_0 (3lx_D^2 - x_D^3)} \right]^{\frac{1}{8}} \quad h = \frac{2L(3lx_L^2 - x_L^3)}{Ev_0 b^3}$$

Again, L and D are the lift and drag force, respectively; E is the modulus of elasticity for the material, l is the total length of the beam, w_0 and v_0 are the deflections in the Z and Y directions, respectively; and x_L and x_D are the positions of the lift and drag sensors, respectively. In this case, L and D correspond to the maximum lift and drag expected for the test airfoil. For maximum angle of attack (30°) and maximum flight velocity (200 ft/sec) the lift and drag, using the flat plate approximation, are expected to be 35 pounds and 7 pounds, respectively.

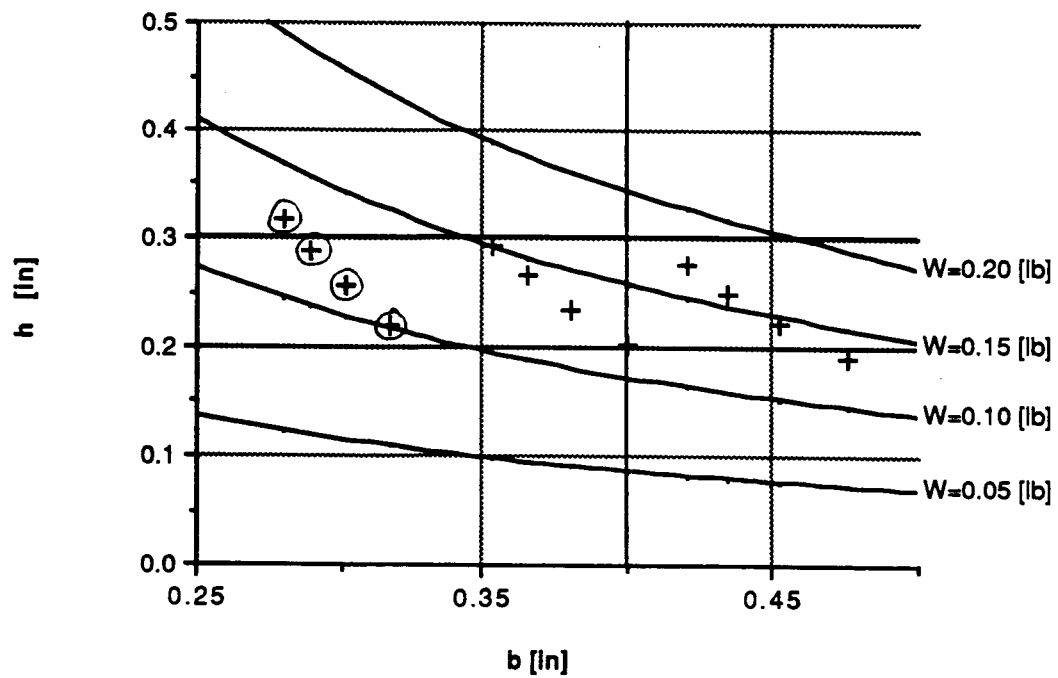
Dimensions for the column were determined based upon different positions for the lift and drag sensor. Its possible failure due to yielding was then checked. Two cases of maximum stress were checked: one at the bottom corner of the column where the bending stresses are highest, and the other at the center of the surface of the longest face where the stress due to torsion is greatest. The shear stresses on the X face of the column were approximated as simply the appropriate force divided by the cross sectional area. Von Mises' yield criteria was used, the column yielding based upon the available yield strengths for the materials.

Graphs of the column dimension data were made for each of the different materials used, and are shown in the following pages. The axis of the graphs are the two dimensions of the column calculated based upon the sensor placements. Curves of constant weight are also indicated on the graph. The data points that are circled indicate the column would yield with these dimensions. Points in the upper right corner past both of the added solid lines are those that satisfy the design criteria. Examining the graphs for steel and titanium, it is easily observed that none of the data points fall in this region, and in general, the titanium configurations are lighter than the those for steel. The aluminum configurations are lighter still, and one data point falls in the desired upper-righthand quadrant. Lightest of all are the magnesium points, and a number of them fall in the correct quadrant. So, the magnesium metal would seem to be the best choice in terms of the merit parameter, weight.

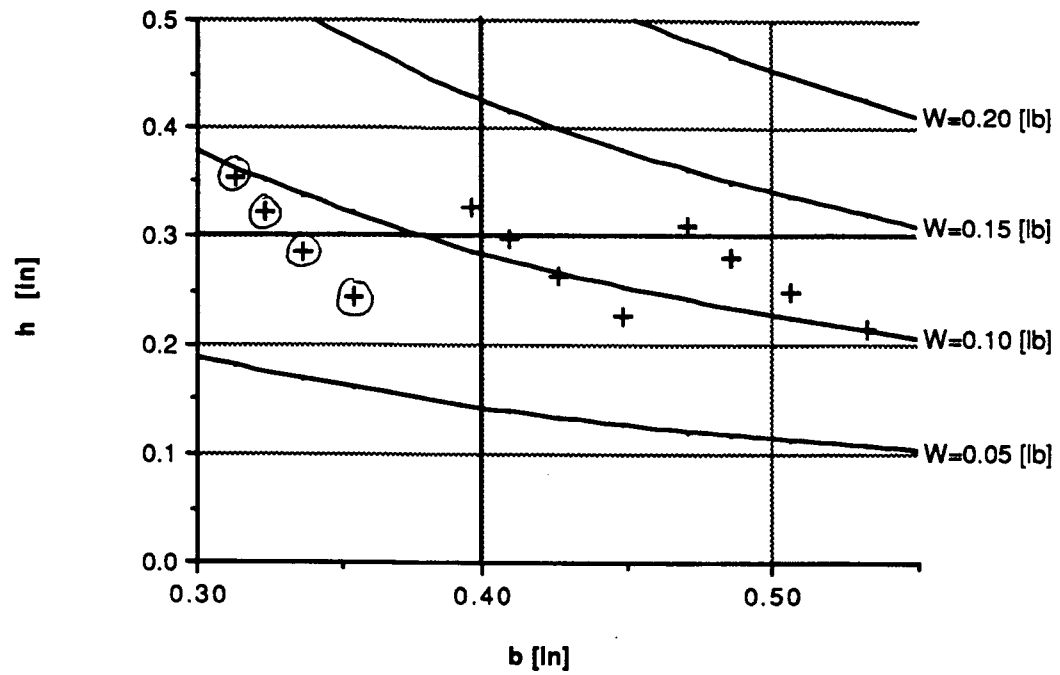
Weight for Steel Shaft Configurations



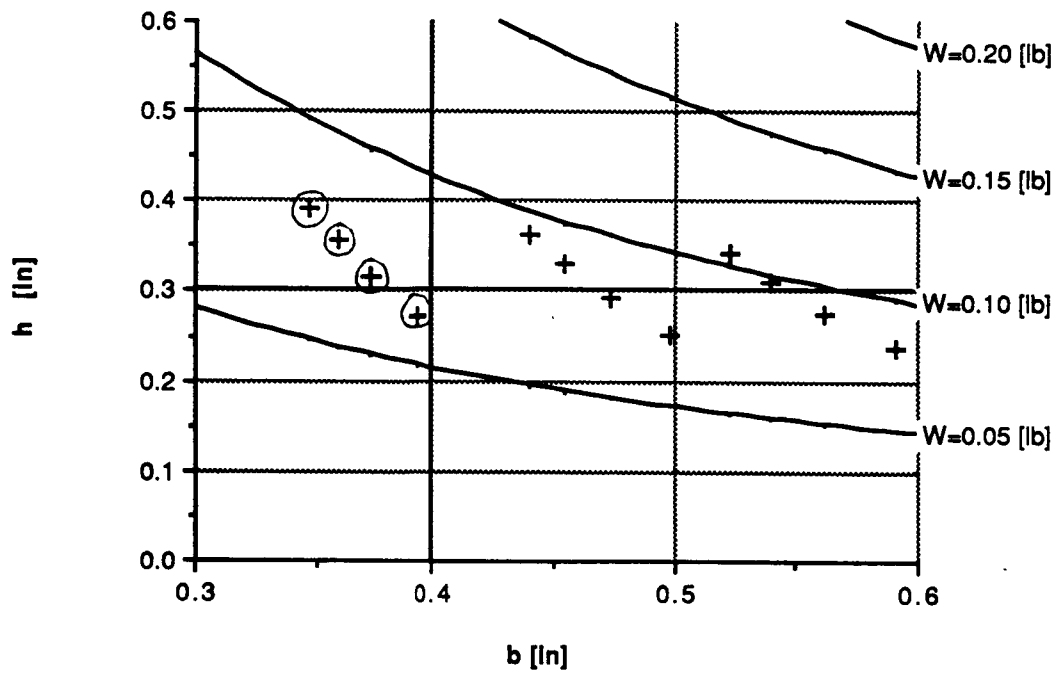
Weight for Titanium Shaft Configurations



Weight for Aluminum Shaft Configurations



Weight for Magnesium Shaft Configurations



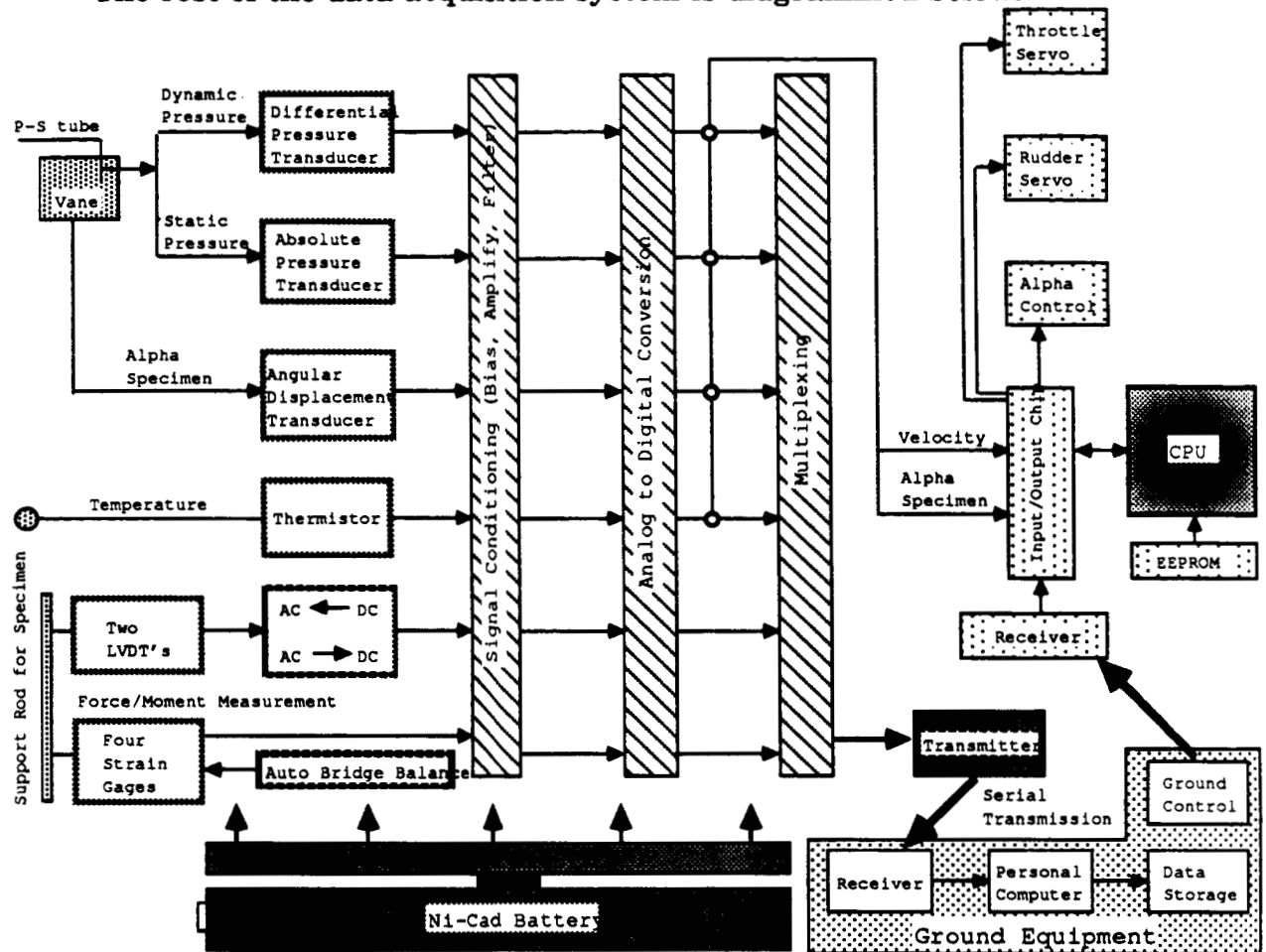
None of the points on the magnesium graph fall particularly close to the intersection of the minimum dimension lines, which would be the case of least weight. More calculations were performed picking dimensions much closer to the limits so that the lightest possible shaft could be obtained that would not fail. Choosing a case where the dimensions would cause failure at the maximum force conditions would not be prudent because the values calculated for the maximum forces could be off. Also, over time the metal may fatigue, which would significantly lower the yield strength of the material. Therefore, a safe, but still light column would be when the ratio of Von Mises' results over the yield criteria is 0.50. This occurs when the dimensions for the beam are 0.45 in x 0.35 in x 9.0 in, and the lift and drag sensors are placed at 3.55 inches and 6.63 inches from the base, respectively. The weight for the column would then be 0.092 pounds.

The choice for the LVDT to use is fairly simple. As stated previously, the column is designed for a maximum deflection of 0.1 inches. This would be the total movement for the drag sensor, and doubling it gives the total movement for the lift sensor. This determines the total range for the two LVDT's, which determines the particular instrument to be used. LVDT's are analog instruments, and therefore have an "infinite" resolution, so theoretically very minute forces could be measured. As far as measuring the moment, strain gages can be purchased in a whole range of sensitivities. It is difficult to determine the exact state of strain in a beam such as this, but using four strain gages of moderate sensitivity should give good results because of the quadrupled sensitivity from using the four in a Wheatstone bridge.

There are a number of possible problems that need to be examined before the force balance suggested is built. Vibrations from the plane's engine could adversely affect the LVDT's, causing them to oscillate instead of giving a steady reading. Twist of the column from the induced moment could also affect the lift and drag readings, but it would seem that if proper calibrations are done before the flight, this should not be that much of a problem. Also, the test airfoil could begin to oscillate wildly at its natural frequency at some point during a flight, which might damage the force balance setup.

Other Instrumentation

The rest of the data acquisition system is diagrammed below:



Besides the force balance setup, there are a number of other instruments to collect data needed for the experimental analysis.

Thermistor — this would measure the air temperature during the flight to within 0.25 degrees Celsius of the true temperature. This data is necessary for determining the velocity, and hence Reynolds number, at which the plane is flying. An error of 0.25 °C would cause a difference of about 0.1 ft/s in the velocity measurement, which is acceptable. The thermistor could just be attached to the surface of the plane with little regard for whether it becomes a stagnation point, because there is very little difference between

the stagnation temperature and the static temperature for the low velocities at which the plane will be flying.

Absolute Pressure Transducer — this would measure the absolute pressure of the air during the flight to within 0.1 psi. This pressure reading is also necessary for calculating the flight velocity. This type of accuracy is available in commercial pressure transducers, and would cause an error in the velocity of about 0.3 ft/s. The absolute pressure would be obtained from the static pressure port of a Pitot-static tube directed into the flow over the plane.

Differential Pressure Transducer — this would measure the dynamic pressure of air during the flight to within 0.005 psi. This is the final reading needed to accurately determine the flight velocity, and would cause an error in the velocity of about 0.8 ft/s. Accounting for the error in all three instruments results in an error in the velocity measurement of about 1 ft/s, which is quite acceptable for experimental testing. The dynamic pressure would come from the same Pitot-static tube used for the absolute pressure measurement.

Angular Displacement Transducer (ADT) — this would be attached to a small vane, which would orient itself on the direction of the flow over the plane. The transducer would measure the angle (from the centerline of the plane) to which the vane had turned, which will be fed into an automatic control system that will maintain the test airfoil at a particular angle of attack during data collection. The same vane will also have the Pitot-static tube mounted on it, thereby keeping it pointing directly into the oncoming air. To avoid interference with the test airfoil, the vane will be mounted vertically on the bottom of the plane. Since it will be an only inch or two long, it will not interfere with landing or be subject to ground forces.

Signals from all the different instruments mentioned would be fed into a signal conditioning circuit, which would remove any bias, amplify them, and remove any extraneous noise. The signals would then be fed into a 16 bit analog to digital converter. This level of A to D converter would provide 65536 different levels for the signals to be converted into, which for

the small voltage output range of these instruments (typically 5 to 10 volts), should provide sufficient accuracy. Also, these A to D converters should be readily available because they are the type used in compact disc players, something becoming quite commonplace. The digital signals would then be fed into a multiplexer for transmission to the ground. Transmission would be in real-time and digital, thereby avoiding most interference problems.

The signal would be received on the ground by a special circuit board plugged into the slot of a personal computer, such as an IBM PC (laptop or portable), Macintosh II, or Apple II. Circuitry on the board would demultiplex the signal, and then convert all the signals from the different instruments into useful values such as pressure, angle of attack, lift force, airspeed, etc. These quantities could then be viewed on the screen in real-time, and at the same time stored to disk for later analysis and graphing. The computer would be in a building near to the flight area, or if the flight area is remote, placed in the vehicle required to get to the remote area, and powered from its battery. The transmitter in the plane would have a range at least as great as the ground radio controlling the plane, generally a mile or two. The flight plan for the Air Rhino remains well within this range.

Special Systems

In order for the data acquisition system to obtain accurate, useful data, a special feedback control system will be used. This is seen on the far right of the diagram in the *Other Instrumentation* section. The person on the ground will transmit a special numeric code that will correspond to a particular flight speed and angle of attack for the test airfoil. This code will be received by a radio receiver on the plane, passed through an input/output chip to a central processing unit (CPU), and then decoded based upon information stored in the EEPROM connected to the CPU. The CPU will then send a signal to the stepper motor beneath the test airfoil to step to the desired position.

Meanwhile, flight airspeed and airfoil angle of attack data would be feeding into the CPU as it is being collected by the different instruments. Thus, the CPU would have the actual values for the airspeed and angle of attack, as well as the desired values, and would then act as a feedback control system. The CPU will send signals to the throttle to attain the desired velocity, and signals to the rudder to attain the desired angle of

attack. Even though the test airfoil will have already been "stepped" to the desired angle (with reference to the plane centerline), the flow over the airfoil will not necessarily be giving the right angle of attack. Knowing the stepped angle and the angle indicated by the ADT, the CPU will be able to determine the actual angle of attack, and adjust the rudder accordingly. A separate dynamic control system (discussed in the *Control Systems* section) will maintain steady, level flight with the ailerons and horizontal stabilizers, so the throttle and rudder are the only controls the CPU must actuate. Once enough data has been obtained, the operator simply transmits another code to regain control of the plane. The plane can then be turned around for another test run.

The electronics part of the data acquisition system would be on two 4 x 6 inch plastic cards surrounded by a vented, metal enclosure onboard the aircraft. Each of the different instruments would simply plug into the "box." All calibration of the instruments would be done automatically by special circuitry on the cards, removing the burden from the operator. Also, there would be a voltage regulator on the cards that would automatically feed the instruments the correct excitation voltages for operation, as well as an DC to AC and AC to DC converter to power the LVDT's. The whole system is estimated to require approximately 350 mA of current, which would be supplied by a 1200 mah ni-cad battery pack.

Weight Estimation

Weight Percentages

The weight estimation for the Air Rhino was determined by first estimating the size and weight of the propulsion, controls, and data acquisition systems. Once these weights were determined, a general idea for the size and weight of the aircraft structure could be determined. The first estimate for the propulsion system was 13.9 lbs, 7.9 lbs for the engine, and 6 lbs of fuel. The data acquisition system weight was originally estimated to be 6 lbs. The force balance was calculated to weigh 2 lbs.; the test airfoil was estimated to add another 1 lb to the weight. The telemetry system on board the aircraft was determined to weigh 1 lb and the battery to support this system was determined to be 2 lbs, based on the weight of the ni-cad battery packs used as the power source for the technology demonstrator. The 11 servos needed for the RPV were estimated to weigh approximately 1 lb. The autopilot added another 1 lb to the controls weight, for a total of 2 lbs of controls.

Once the weights for the propulsion, control, and data acquisition systems were estimated, the actual weight of the RPV structure was determined. Data on weight percentages for working aircraft were found in existing literature.¹ From this data, the actual structural weight for aircraft was determined to be approximately 38% of the total weight. Using this percentage, and knowing the total weight of the RPV minus the actual structural weight, the actual structural part of the RPV was calculated to be 13.4 lbs. From this structural weight, the weight of the various structural components could be determined. Of the 13.4 lbs for the structural system, the wing would weigh 5.4 lbs and the fuselage and tail were estimated to be 5 lbs and 3 lbs, respectively. Based on the total weights of the component systems, the total weight of the RPV is 35.3 lbs. This allowed 39% of the total weight for the propulsion system, 17% for the data acquisition system, and 6% for the control system. The weight percentages and estimated weight for each component system are summarized below:

¹From several RPV and model airplane magazines, and from talking to our 'resident expert' on RPV systems, Mr. Joe Mergen.

WEIGHT PERCENTAGE ESTIMATES		
Component System	Weight Percentage	Component Weight (lbs.)
Structural	38%	13.4
Propulsion	39%	13.9
Data Acquisition	17%	6.0
Controls	6%	2.0
Total Component Weight:		35.3

Center of Gravity Estimation

The calculation of the center of gravity was more complex than the estimation of the component weights and the weight percentages. The weight of each general system had to be broken up into its several components and the location of each of these components had to be known in order to calculate the center of gravity location. Also, continual iterations of weight estimates kept changing the center of gravity calculations.

For the final center of gravity calculation, the weight of fuel needed for our engine was changed to 2 lbs. from an initial estimate of 6 lbs, due to a lower than expected specific fuel consumption. This weight reduction is rather drastic, approximately 17%. With this reduction in weight, the structural weight could have been reduced slightly, but was kept constant to allow for future expansion of payload. The payload weight could be increased if more fuel was added to increase the RPV's endurance. The payload weight could also increase by the installment of additional data acquisition equipment, or a more sophisticated telemetry system.

After the initial estimates of weights were refined through the design process, the center of gravity was found using locations and weight figures for each component of the Air Rhino. These component weight figures are more accurate than the initial weight figures, since the initial figures were based largely on weight percentages and preliminary gross estimates.

A drawing is given below to show the placement of each of these components for the entire aircraft, and a table summarizing the majority of the RPV's component weights and locations follows. The component center of gravity location is referenced from the nose of the RPV.

CENTER OF GRAVITY LOCATIONS		
Component	Center Gravity location (ft)	Weight (lbs.)
Fuselage hull	1.85	5.0
Booms	4.85	1.4
Horiz. stabilizer	6.25	1.2
Vert. stabilizer	6.35	0.4
Battery	0.583	1.5
Autopilot	1.0	1.0
Test airfoil & force balance	0.167	2.5
Circuit Boards & instruments	1.67	0.5
Wing	2.675	5.4
Fuel	3.0	2.0
Engine & Propeller	4.0	<u>8.0</u>
total accountable weight:		28.9

The following equations were used to calculate the center of gravity:

$$X_{CG_{from\ nose}} = \frac{\sum X_{CG_{comp}} \times Wt_{comp}}{Wt_{tot}}$$

$$X_{CG_{from\ nose}} = \frac{(68.13 + 2.5 \times l_s + 5.4 \times l_w)}{28.9}$$

$$X_{CG_{from\ L.E.}} = X_{CG_{from\ nose}} - l_w + \frac{\bar{c}}{4}$$

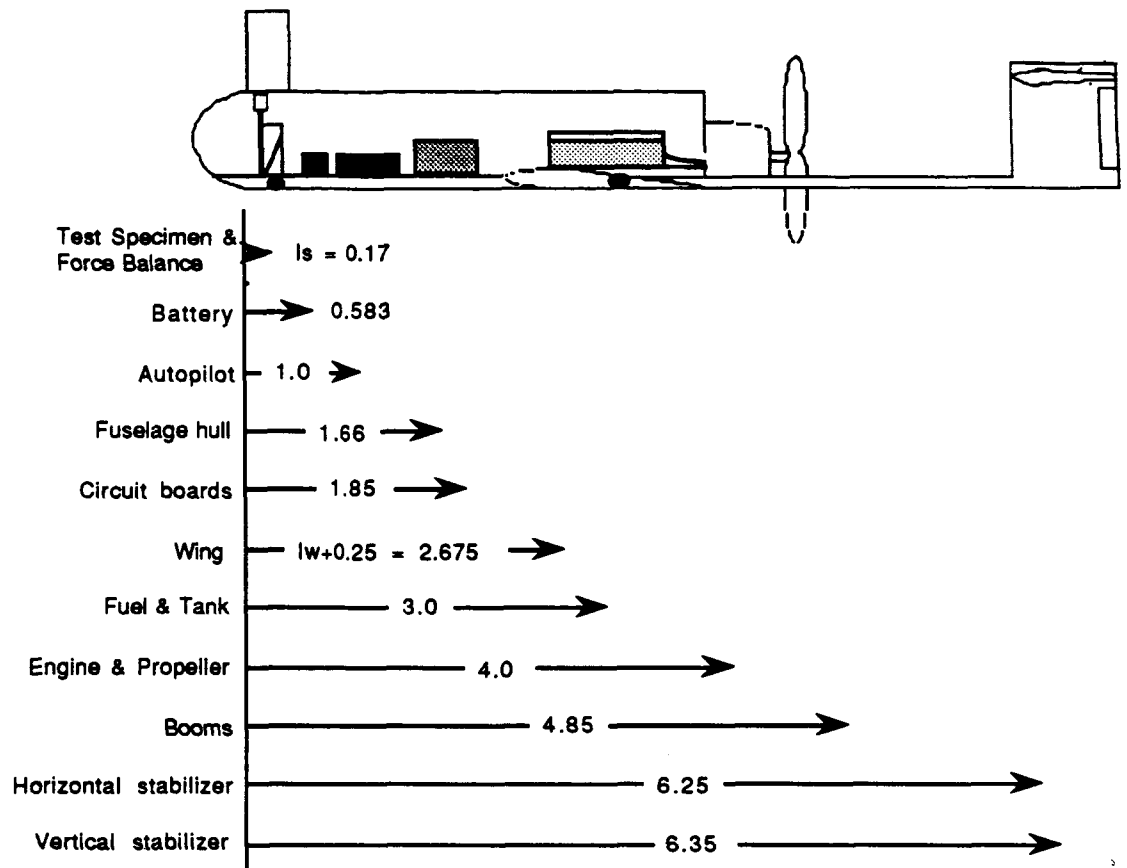
where,

$$l_s = 0.167 \text{ ft.}$$

$$l_w = 2.425 \text{ ft.}$$

$$c = 1.5 \text{ ft.}$$

$$\text{Therefore, } X_{CG_{from\ L.E.}} / c = 0.517$$



The values used in the center of gravity calculation have a significant amount of error associated with them. The largest error is associated with the RPV's structural components. The weight of each structural component is based on guesses of the approximate percentage of the total structure weight that would be constituted by each of the components. These guesses are based primarily on the size of each component, but the relative sizes of structural elements does not necessarily equal their relative weights. Also, the component center of gravity location was determined easily for regular symmetric shapes, but was more difficult for more

irregular shapes. The center of gravity location for airfoil sections, for example, was estimated to be approximately halfway between the quarter-chord point and the mid-chord.

The total weight for the center of gravity estimation adds up to only 28.9 lbs. This value for the weight falls 6.4 lbs below the previous estimate of 35.3 lbs which was determined in the weight percentages section. From the list of components used in the center of gravity calculation, it is obvious that some components were not included. They were omitted because either their weight, or their location, or both, were difficult to determine. A brief list of extra components, with a rough estimation of their weights is included below:

EXTRA COMPONENT WEIGHTS	
Extra Components	Est. Weight (lbs)
Servos	1.0
Control linkages	0.8
Retractable landing gear	1.0
Metal plate for fuselage floor	2.0
Securing devices and electrical devices	1.6
total	6.4
total accountable weight (from table above)	28.9
total plane weight	35.3

These "unknowns" constitute the extra 6.4 lbs that is neglected from the center of gravity calculation. It is thought that these extra components will shift the center of gravity location, but not significantly. This is because many of these components are located near the center of gravity location. A more accurate calculation of the location for the center of gravity should take these extra components into consideration.

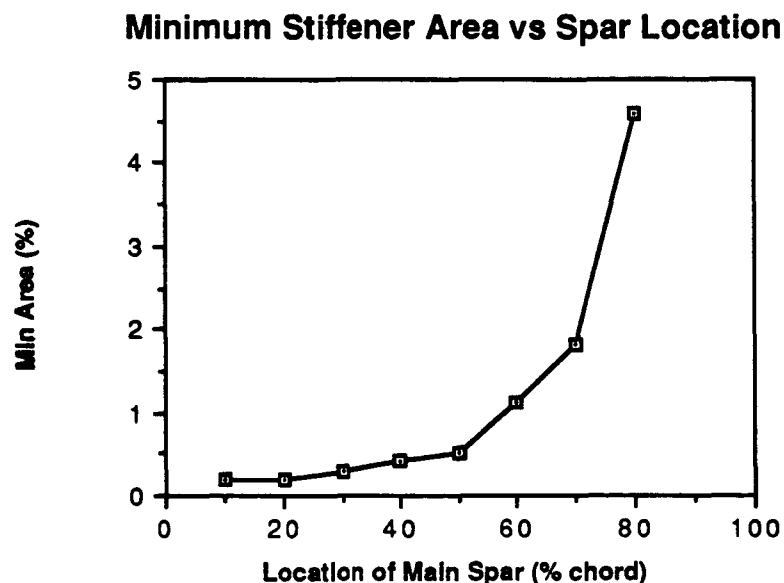
Aerodynamic Design

Wing Design

Wing area was one of the first parameters determined for the Air Rhino. The wing area is chosen on the basis of the desired wing loading for the airplane's mission. Most RPVs in the weight class of the Air Rhino carry wing loadings between 1.6 and 3.0 lb/sq ft. Given the estimated plane weight of 35.3 lb, a wing area of 15 sq ft was chosen to yield a wing loading of 2.35 lb/sq ft. The aspect ratio of 6.33 was also chosen on the basis of RPVs in the Air Rhino's class, and also to ease construction by providing a reasonable chord length and span.

Structural design of the wing was aided by a computer program which calculated the optimum placement location of the main spar in the wing structure. The factors determining the optimum placement were the minimum required stiffener area to withstand the stresses produced by a certain loading and the corresponding wing structure weight associated with that area.

Graphs produced from the data generated by the program are included below:

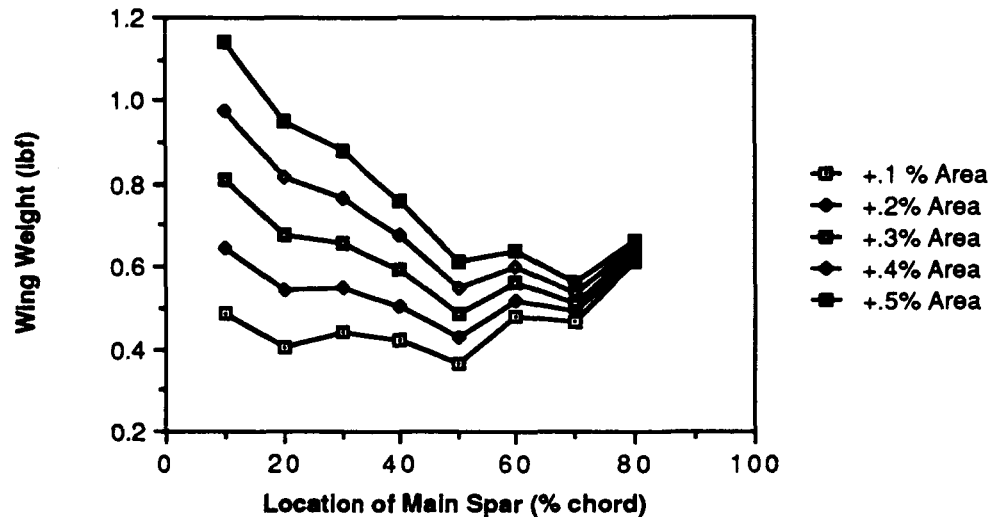


This first graph shows the stiffener area percentages required as the

spar is located at points along the chord of the wing. The area remains fairly constant until the spar is aft of the 50% chord location, then rapidly grows.

The relation of the spar location to the wing structure weight can be calculated with a formula involving the wing area directly. Thus the weight of the structure can be exactly correlated to the area of the structure. A graph illustrating this relationship is shown below:

Wing Weight vs Spar location and Differential Area



The minimum wing weight for stiffener areas of up to 5% of a given wing area falls at 50% of the wing chord. From these two graphs it appears that placement of the main spar at the 50% chord position would strike a compromise between the minimum wing weight condition and the minimum stiffener area required. This would be useful in the proposed lightweight design.

The wing will be constructed with a rib and spar structure, with smaller support spars placed along the wing to provide torsional rigidity for the structure. This construction is one of the easiest and lightest methods for wing building. Materials for this section will probably be mostly aluminum, a material that is readily available, easily workable, and inexpensive. Spruce wood might also be used. Materials choice is more fully discussed in the Materials Selection section (q.v.).

A layer of "Monokote" or equivalent mylar seal coating would cover the rib and spar structure to act as an aerodynamic skin. Extra strength

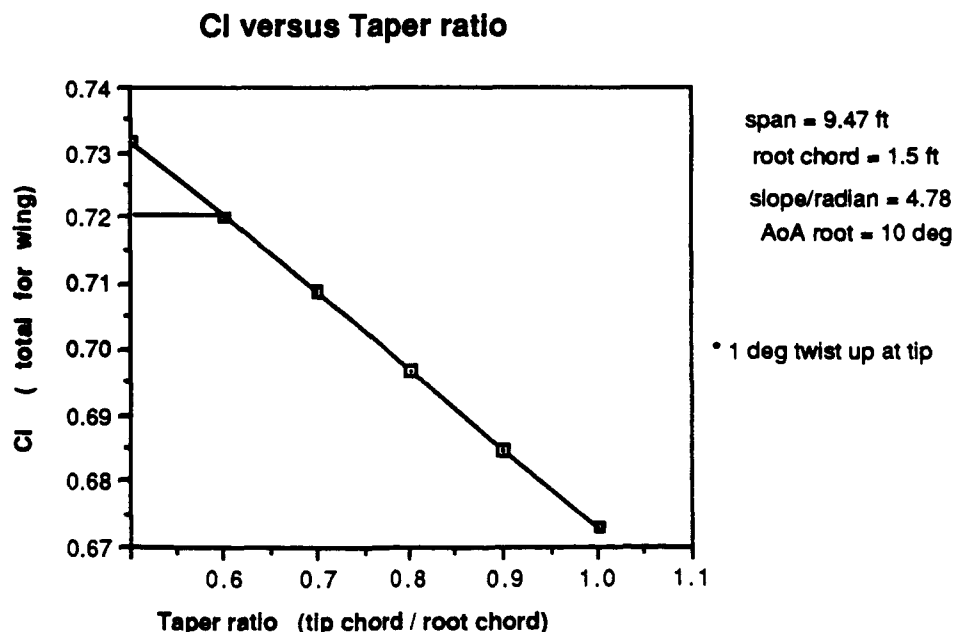
could be added by applying a sheeting layer of thin balsa on the wing before covering with the "Monokote", but experience with the technology demonstrator has shown that mylar should be sufficient.

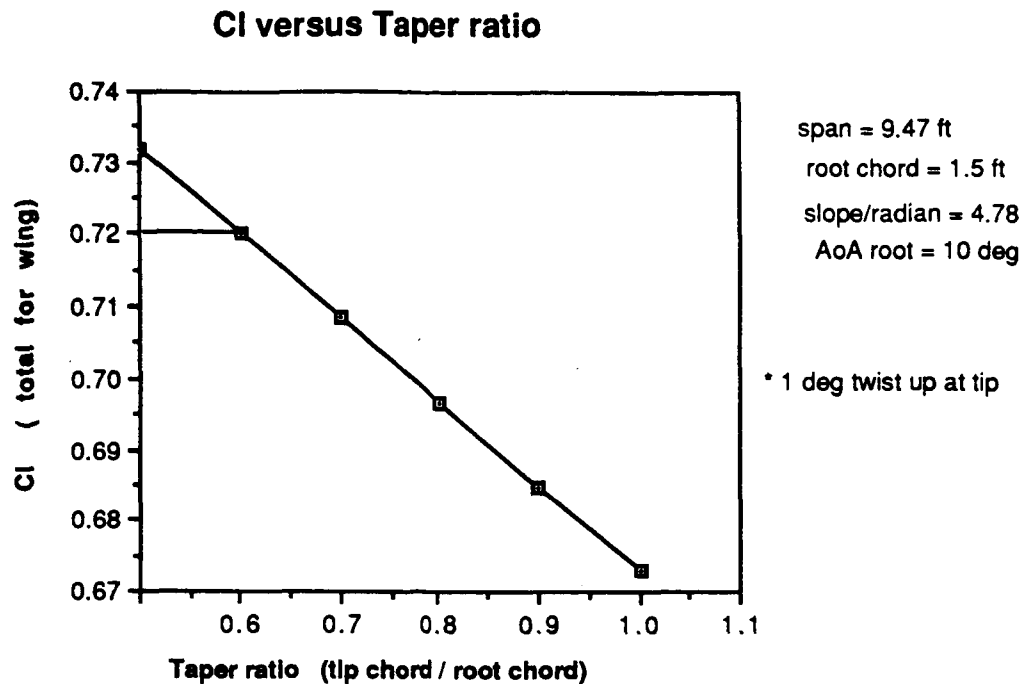
The wing would be in two sections: an inboard and an outboard section. The inboard part would be connected directly to the fuselage structure, and be structurally reinforced to provide support for the twin tail booms. This section has no dihedral. The outboard section would be at a dihedral of about 5° for control purposes. Since this section does not need to carry the tail boom loads, the structure and weight would be much less than that of the inboard section.

Airfoil Selection

The airfoil shape for the wing was chosen to be the Wortmann FX 63-137. It is known for its favorable characteristics in the low Reynolds number ranges and high stall angle with a relatively high maximum lift coefficient. The Wortmann airfoil has a slope/radian value of 4.78, determined from the lift curve. From studies done on the wing lift distribution and lift coefficient values, a tapered shape with a small twist at the tips, both of which boost lift, are required to achieve a suitable maximum lift for the landing and takeoff speeds of the aircraft.

The details of the wing were determined from the graphs below:





A taper ratio of 0.6 was chosen for a wing of 9.47 ft span and 1.5 ft root chord. The root chord was later changed to 1.875 ft, and the span to 9.73 ft but the taper ratio was kept at 0.6. The Wortmann airfoil has a slope/radian value of 4.78, determined from the lift curve. A root angle of attack of 10° at takeoff and landing is required to achieve a lift coefficient of .72027, the maximum available based on lifting line theory calculations.

Fuselage Design

The fuselage shape was designed to produce as little drag as possible while containing volume necessary for instrumentation, controls and propulsion. A relatively flat fuselage, from the top at least, was desired on which to mount the test airfoil. A cylindrical fuselage creates construction challenges when installing the internal elements of the fuselage but incurs low drag. A fuselage with a rectangular cross section would be easier to construct, but produces more drag.

The cylindrical fuselage was eventually chosen, with a mounting plate added. A thin metal plate can be installed approximately two inches from the bottom of the fuselage to eliminate installation problems with the sensitive instruments, such as the force balance. This plate acts not only as firm structural support for the balance, but it also provides a more

convenient flat floor to mount the other internal elements of the fuselage, such as the heavy battery. Other lighter elements could be mounted and secured using a lighter material than metal. The diameter of the fuselage was chosen to be 9 in, primarily to accomodate the engine and engine mount.

The front of the fuselage should not be blunt, because a rounded nose will produce less drag. A long, slender nose will produce less drag than a short, stubby nose, but a long nose will also increase the amount of material needed for fuselage construction and therefore increase the weight of the RPV. A compromise in nose taper ratio is something that should be studied in more depth. It was also found that less wetted area would be generated by a sharply pointed cone-shaped nose than by a rounded nose, like in the design of a bullet.¹ The reader is advised that this is a start for deciding upon a design for the nose. Further research into the nose construction still needs to be considered.

The back end of the fuselage will consist of a cowl to reduce the drag over the engine. The design of this cowl is discussed in more detail in the Propulsion section (q.v.).

Interference on the Test Airfoil

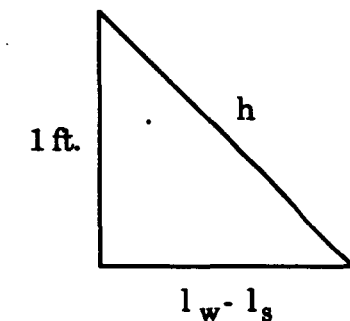
The installation of the test airfoil on the front top surface of the fuselage creates some air flow interference. The flow conditions on the test airfoil must be taken into consideration when conducting experiments. Interference can cause data to differ from actual characteristics if the type(s) of interference are not part of the process of interpreting the experimental results.

The effect the propeller has on the flow over the test airfoil should be examined. This effect is difficult to estimate and needs more consideration, but by pushing the test airfoil as far as possible to the front of the fuselage, this effect is at least minimized.

The determination of the induced velocity on the test airfoil created by the upwash from the wing was roughly determined. One would expect the upwash on the test airfoil to decrease as the distance between the wing and

¹Much of the fuselage nose design material from D. Stinton, The Design of the Airplane, Van Nostrand, 1988.

the test airfoil increases. An attempt was made to determine exactly how the upwash varies. The upwash from the wing was calculated by placing a single vortex at the quarter chord location of the wing. This vortex was oriented clockwise (from the view out at the left wingtip) with its centerline oriented wingtip-to-wingtip. The velocity this vortex produces at the midpoint of the test airfoil section was calculated. This velocity is a function of the direct distance between the wing's quarter chord point and the middle of the test airfoil section. The resultant velocity lies in the plane normal to the centerline of the vortex, perpendicular to the line drawn from the wing's quarter chord point to point of application of the test airfoil's drag. The velocity determination is illustrated below:



$$h = \sqrt{1^2 + (l_w - l_s)^2}$$

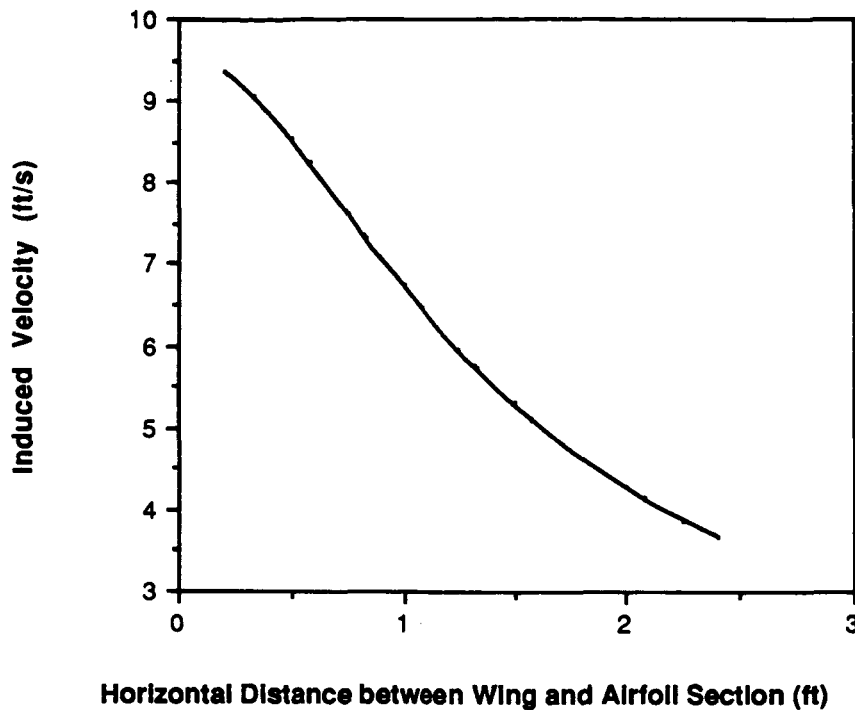
$$V = \frac{\Gamma}{4\pi h}$$

where

$$\Gamma = \frac{L'}{\rho_{\infty} U_{\infty}} = \frac{1}{2} C_L U_{\infty} \bar{c}$$

From the velocity equation it can be seen that the greater the separation of the test airfoil from the wing, the smaller the induced velocity from the wing upwash. This variation is shown in the graph below:

Induced Velocity from Vortex at Wing as a Function of Horizontal Spacing



For the test airfoil mounted at the front of the fuselage and the quarter chord of the wing located 2.425 ft. from the aircraft's nose, the induced velocity on the test airfoil in its established direction is 3.867 ft/s.

The boundary layer thickness is only a function of the distance of the test airfoil from the nose of the RPV. Equations for boundary layer thickness for a flat plate approximation are used; the fuselage of the RPV can be modeled as a flat plate with relatively good accuracy because the majority of the fuselage is flat except for the nose taper region. The distance used in the boundary layer thickness calculation started at the point of the fuselage where its cross sectional area becomes constant. More accurate determinations of the values for the boundary layer thickness could be found from a model which is more true to the actual shape and conditions of the fuselage, but this model will require future study.

The equations for boundary layer thickness were found for both laminar and turbulent flow. For a true flat plate, at a Reynolds number of approximately 5×10^5 , transition from laminar to turbulent occurs. This

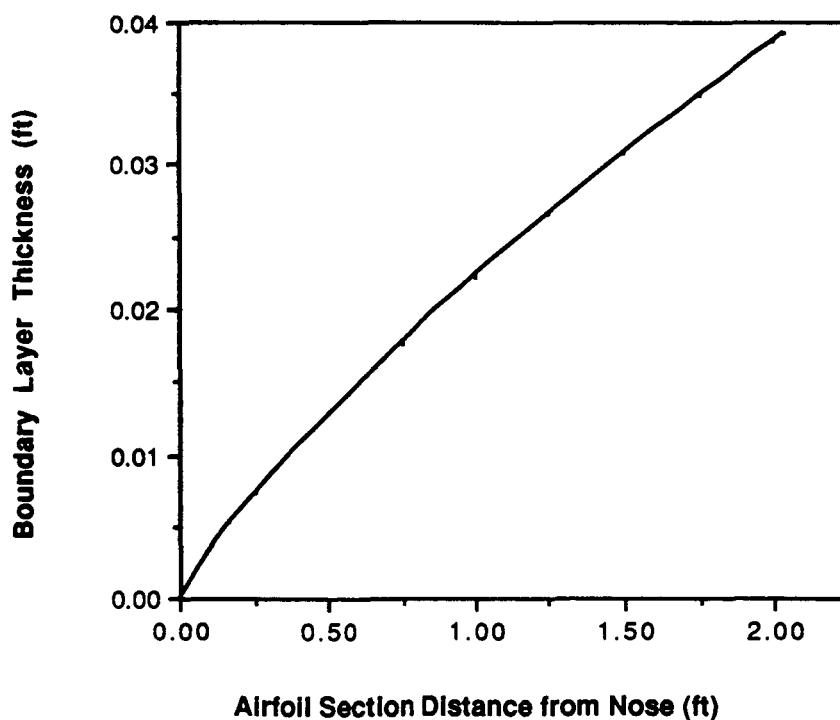
transition would occur approximately 0.40 ft from the start of the cylindrical section on the Air Rhino fuselage. Since the fuselage is not a true flat plate, and since there will probably be some material buildup on the fuselage, flow was modelled as turbulent.

The boundary layer variation with varying location of the test airfoil was examined. The boundary layer thickness for turbulent flow, δ_t , is related to the test airfoil placement from the nose of the RPV by the equation:

$$\delta_t = \frac{0.37 l_s}{\left(\frac{\rho U_\infty l_s}{\mu}\right)^{\frac{1}{5}}}$$

From this equation, the the boundary layer thickness decreases as the test airfoil is moved closer to the nose of the RPV. This is illustrated in the following graph:

Turbulent B.L. Thickness as a function of Airfoil Section Placement



This trend is verified by the graph of boundary layer thickness as a function of test airfoil location which is included. For the wing and test airfoil placement mentioned earlier, the boundary layer thickness is 5.32×10^{-3} ft. (0.064 in.).

The results of this section are summarized in the table below:

STATISTICS ON TEST AIRFOIL INTERFERENCE	
Test Section distance from nose (ft)	0.167
Wing distance from nose (ft)	2.425
Distance between (ft)	2.258
Velocity induced (ft/s)	3.867
Boundary layer thickness (ft)	0.00532

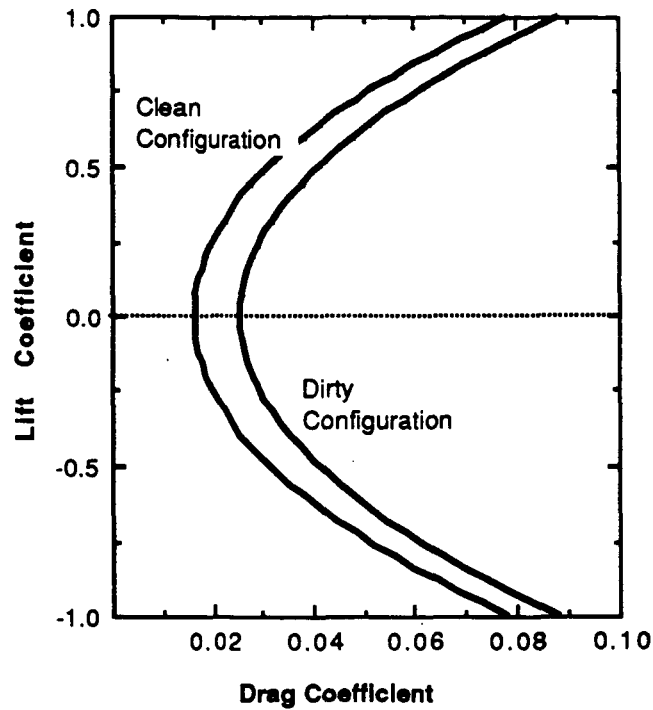
Drag Prediction

The drag forces on the craft were predicted from a drag polar derived for the specific configuration of this aircraft. The parasitic drag value of the drag polar was found from a drag estimation technique that defines a parasitic drag coefficient for each individual component of the aircraft. These component coefficients are multiplied by the component effective area, summed and then divided by the reference area of the Air Rhino.

Because of the design of this plane, some slight modification was required for the above technique. The method above takes the different cases of propeller driven craft and jet driven craft. The propeller consideration assumes that the flow everywhere over the craft is influenced by the propeller wash. The jet case assumes every component to be in relatively undisturbed flow. For the specific craft of this design, a combination of the two cases was used. First the drag coefficient estimations for components up to the pusher propeller use the jet case, as the flow here is relatively undisturbed. Aft of the propeller, the booms and tail surfaces are in the propeller wash and have the coefficients calculated using the propeller case.

The drag polar for the craft is below:

Air Rhino Drag Polar



The predicted total parasitic drag force on the entire aircraft at maximum speed was 1.23 lb. This value then was used in designing the aircraft propulsion system.

Drag forces attributable to the test specimen were calculated, in a worst case scenario of maximum speed of 200 ft/sec and maximum test airfoil deflection at 30°, using the flat plate assumption for the test airfoil. Drag forces of about 7 lb were expected for the specimen. Lift forces were calculated to be about 35 lb.

Propulsion

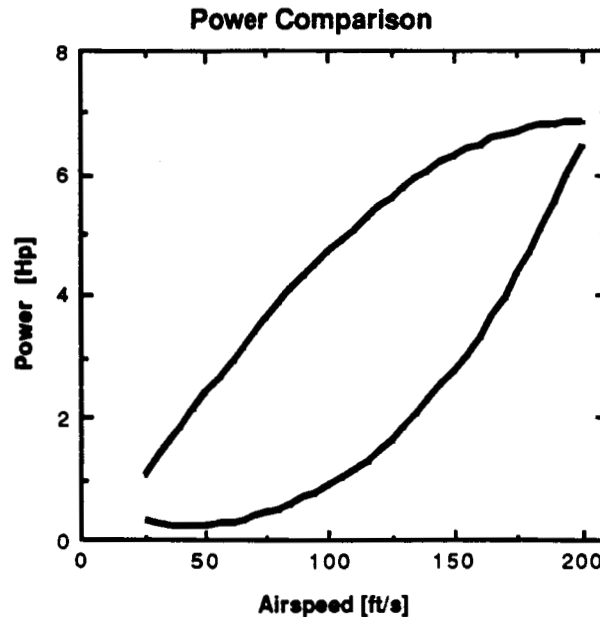
Propulsion System and Airframe Integration

Air Rhino is designed in a 'pusher' configuration that was chosen to avoid disturbances to airflow about the test airfoil. A tractor configuration normally would cause interference in the test airfoil region due to propeller wash and tip swirl effects. In the pusher configuration, the propulsion system would be mounted on the rear of the main fuselage, between the two tail booms and, more importantly, well downstream of the test airfoil.

One concern with this configuration is washout of the tail control surfaces by the propeller wake, reducing stability and control. Control in similar designs, however, has been adequate. Another concern is blockage of flow through the propeller by the 9 in diameter fuselage. To avoid blockage in this design, a long diameter propeller will be used, and a tapered rear fuselage and engine cowling will provide as much laminar flow to the propeller as possible.

Power System Selection

An analysis of power required vs. power available for a target velocity range of 50--200 ft/s resulted in the power comparison curve shown below:



This figure represents the aircraft in a worst case, dirty configuration, with the test airfoil at an angle of attack of 30° to the free stream. In this configuration, the airfoil adds up to 7 lb to the overall drag at $V_{\max}=200$ ft/sec. Accounting for propeller efficiency as a function of airspeed, this plot shows that a maximum shaft power of 8 Hp is necessary to fulfill power requirements at maximum airspeed, allowing for a margin of error of 0.5 Hp.

Both electric and internal combustion engines were considered for supply of power. The electric system has the advantage of cleanliness, because it requires no liquid fuel supply and emits no exhaust. The electric motor also operates at a lower noise level than the reciprocating type. However, no electric propulsion systems were found that could provide 8 Hp. The electric system was therefore removed from consideration.

Preliminary analyses also suggested the possibility of using a ducted fan engine in the design; the argument was that, by using ducted fan blades instead of an open-air propeller, tip vortices could be eliminated and thus wake turbulence reduced. Because of the pusher configuration, the engine wake often passes directly over the tail control surfaces, and if the turbulence due to tip vortices could be eliminated, control of the the tail, and therefore the entire aircraft, would improve.

A ducted fan engine, however, is more complicated, more expensive, and heavier than a comparably powered open-propeller system. As mentioned previously, several pusher RPV designs have proven that successful tail surface control is possible with an open propeller. These arguments led to the selection of a conventional, reciprocating propulsion system for this design.

The specific engine chosen was the Quadra Q-82 two-stroke, gasoline powered engine. This engine can provide up to 8 Hp and 8000 RPM. It uses gasoline as fuel, making it less expensive and cleaner to operate. The Q-82 displaces 5 cu in, has a bore of 2.06 in and a stroke of 1.38 in. In terms of performance range and economic factors, the Quadra Q-82 was the the best choice of available engines for Air Rhino.

Propeller Selection

The Quadra Q-82 is matched with a 16.3 in, three bladed propeller with a 23° blade angle at $3/4$ radial position, designed to provide maximum

efficiency for the 8 Hp engine at V_{\max} , where power required is maximum. This propeller-engine combination produces 18.8 lbs of thrust at V_{\max} , and 21.8 lb of static thrust according to the following equation:¹

$$T_{\text{stat}} [\text{lb}] = \frac{(29000) * S_b \text{Hp}}{n * D} = \frac{(29000) * 8}{8000 * 1.33} = 21.8$$

where n is engine speed [RPM] and D is propeller diameter [ft].

The three bladed propeller serves to reduce the required propeller diameter while maintaining high efficiency. The separation distance between the twin booms of the tail section is directly dependent on the size propeller to be used. A small diameter allows for a small separation distance, reducing the weight of structural supports for the tail, and minimizing overall cross sectional area. Further, common knowledge of propellers dictates that propeller noise levels decrease with increasing number of blades. In other words, three blades is quieter than two.

This reasoning would therefore seem to suggest even a four bladed propeller, in order to further decrease the necessary diameter. However, four bladed propellers are more expensive than the three bladed type, and it was found that, at V_{\max} and maximum engine speed, the the extra blade decreases propeller efficiency by 1%. The four blade efficiency drops off more sharply with decreasing airspeed. Further, the flow blockage effect due to the fuselage, mentioned above, would be more significant for four blades than for three, due to the decreased diameter. Because of these problems, the four blade prop was dropped from consideration.

It was found that the type of propeller airfoil section had little effect on efficiency and thrust above an airspeed of 30 ft/s.² As a matter of course, however, a Clark Y section was chosen because, although less efficient at takeoff, it possesses low minimum drag and high efficiency at cruise airspeed and above.³

Propeller thickness and chord length distributions have a more significant effect on efficiency. Propeller performance is highly influenced by Reynolds number, which varies from root to tip sections. The optimum

¹Falk, Karl H., Aircraft Propeller Handbook, Ronald Press, New York, 1937, p. 51.

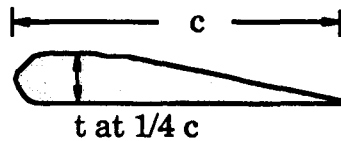
²Joe Mergen.

³Falk, Karl H., Aircraft Propeller Handbook, Ronald Press, New York, 1937, p. 15.

blade section varies from cambered thick plates on the inboard sections to thin tip sections. An optimum propeller would have controllable thickness distribution, varying with changes in engine speed. This type of system would be very complex, so a moderate mean thickness which works well over a wide range of engine speeds would be the most practical.

Thickness and chord lengths for various stations on the propeller were determined from Tables found in Falk.¹ The results are shown in the table below:

Propeller Blade Chord and Thickness Distribution



$R = 8.16 \text{ in}$

Station [in]	r/R	c/D	c [in]	t/c	t [in]
4.08	0.5	0.072	1.18	1.1	1.29
4.90	0.6	0.072	1.18	0.9	1.06
5.71	0.7	0.068	1.11	0.8	0.89
6.53	0.8	0.059	0.96	0.7	0.67
7.34	0.9	0.046	0.75	0.6	0.45

It can be seen that this propeller design is very thick, particularly halfway between root and tip, where thickness, t , is greater than chord length, c . This type of blade has been shown to work well over a wide range of engine speeds.² Another reason for a thick blade distribution is that the power coefficient increases with blade width, but the efficiency goes down due to increased slipstream velocity with the wide blades. However, the efficiency variation between the thinnest and thickest propellers is only 1%.³

Engine Speed Control

The Quadra Q-82 is fitted with a servo arm throttle, which can control engine speed from idle to full speed. The servo arm would run from

¹Falk, Karl H., Aircraft Propeller Handbook, Ronald Press, New York, 1937, p. 51.

²Falk, Karl H., Aircraft Propeller Handbook, Ronald Press, New York, 1937, p. 109-113.

³Falk, Karl H., Aircraft Propeller Handbook, Ronald Press, New York, 1937, p. 109-113.

the remote control servo cluster, in the fuselage just aft of the wing, to the throttle mounted on the side of the engine. This mechanism makes engine speed control fully remote, whether manually or through a feedback controller.

Cowling for Drag Reduction

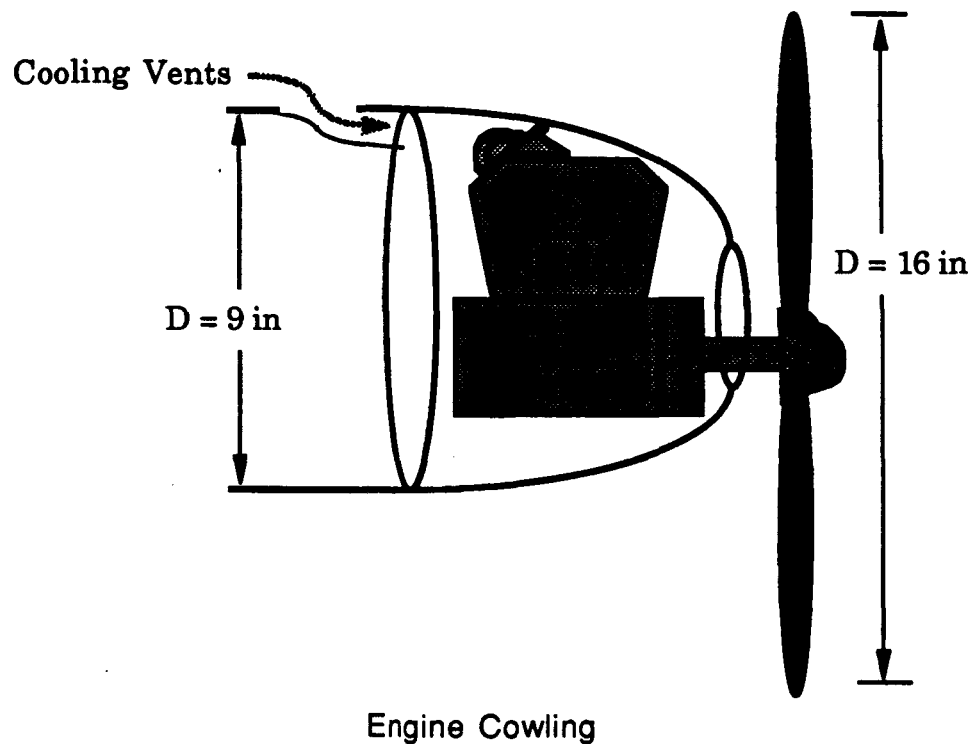
Based on drag coefficients taken from Fluid-Dynamic Drag¹ it was determined that fitting a cowling to the propulsion system would reduce Air Rhino's profile drag by 10%. Though the cowling has not been completely designed, there are some features which it must possess.

The cowling must have a high degree of taper, to reduce blockage of flow through the propeller due to the large, 9 in diameter fuselage. A tapered rear fuselage cowling will increase the laminar nature of the flow around the rear of the fuselage, increasing the flow to the propeller. An ellipsoidal shape like the one illustrated below probably would perform well.

Air Cooling

The cowling must also be vented to provide adequate air cooling to the Q-82 engine. Due to lack of information regarding the cooling performance of the Q-82's finned cylinder, no quantitative analysis could be conducted to determine the requirements for adequate cooling, such as inlet area, contraction ratio, etc. However, a trade study of finned cylinders of this size indicated that the engine may not receive adequate ventilation when Air Rhino is flying near V_{stall} . This could become a concern if a large number of tests in that airspeed range are conducted consecutively. Short period flight at V_{stall} , such as during takeoff and landing, should be no problem.

¹Sighard F. Hoerner, Fluid-Dynamic Drag, published by author, Midland Park NJ, Chapter 13.



Fuel Efficiency

Air Rhino's propulsion system is designed for a minimum test session endurance of 30 minutes. Because fuel consumption is a function of airspeed through its related engine power requirements, the fuel requirements for a 30 minute flight are dependent on the angle of attack of the test airfoil, and the velocity at which each test will be run. Air Rhino was designed for an endurance of 30 minutes at maximum test velocity of 200 feet per second, with the test airfoil at maximum angle of attack. Because its Q-82 engine a low fuel consumption rate of 1.3 lb/Hp/hr gasoline, Air Rhino can achieve this goal using only 39 ounces (1.82 lb) of gas. At 55 ft/s, which is its airspeed for maximum fuel efficiency, Air Rhino run tests for up to 101 minutes, more than 1.6 hours, with the test airfoil at maximum deflection.

For more information on Air Rhino's endurance, see the Range and Endurance section.

Control Systems

Static Margin and Neutral Point Analysis

The static stability of the Air Rhino is primarily dependent on the location of the RPV's stick fixed neutral point with respect to the center of gravity location. Basically, the center of gravity must be located in front of the stick fixed neutral point location.

Two different analyses were used to calculate the static margin. The first method, based on target location of the neutral point, was used as the basis for most of the control sizing and location. The second method is more involved, but probably more accurate. As design iterations of the Air Rhino progress closer to production, the results of the second method should probably be used.

In the first analysis, the target stick fixed neutral point location of the RPV was set to at least 0.5 times the wing mean chord length measured from the wing's leading edge. With the RPV's center of gravity location no further back than 0.3 times the wing mean chord length measured from the leading edge, this yielded a target static margin of 0.2 times the mean chord length. This static margin was four times the recommended value for a full scale aircraft of 0.05 times the mean chord.¹ The target static margin was still used, though, because differences between a full scale airplane and an RPV account for the difference in static margin values.

In the second analysis, the static margin is determined from actual center of gravity and neutral point calculations. The static margin is still defined as:

$$SM = \frac{X_{NP}}{\bar{c}} - \frac{X_{CG}}{\bar{c}}$$

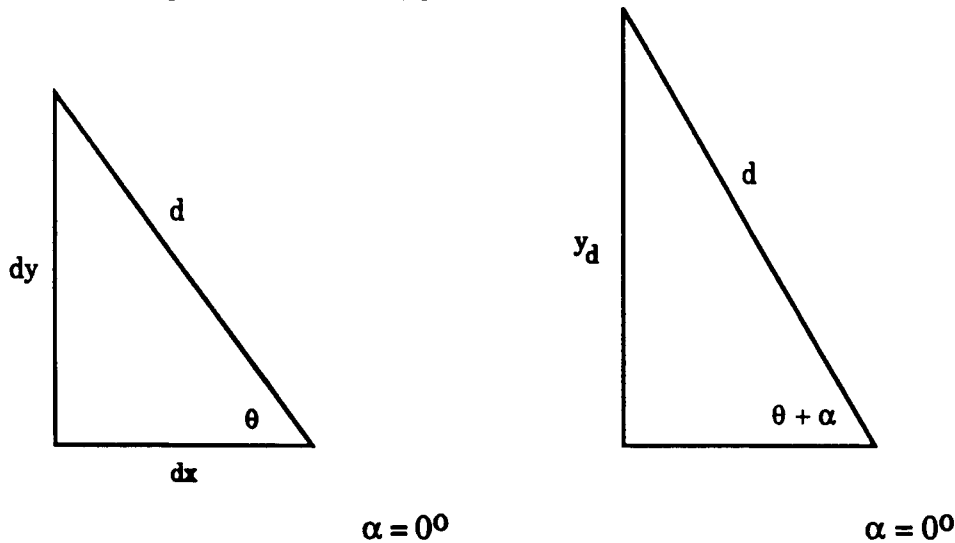
The equation for $C_{m\alpha}$ for the the aircraft had to be known in order to develop an equation for the neutral point. A neutral point equation is found by setting the $C_{m\alpha}$ equation to zero and solving for the center of gravity. An equation for $C_{m\alpha}$ was found to be as given below:²

¹Nelson, Robert C. Flight Stability and Automatic Control, McGraw-Hill, New York ,1989.

²Nelson, Robert C., Flight Stability and Automatic Control, McGraw-Hill, New York,1989, p. 52.

$$C_{m\alpha} = C_{l_{\alpha}} \left(\frac{X_{CG}}{\bar{c}} - \frac{X_{ac}}{\bar{c}} \right) + C_{m_{\alpha}} - \eta V_H C_{l_{\alpha}} \left(1 - \frac{d\epsilon}{d\alpha} \right)$$

There were two problems with this equation that did not allow it to be applied directly to the RPV. (The target analysis used this equation directly to set a target neutral point). The primary problem is that this equation does not include a term to take into consideration the moment contributed by the test airfoil. This moment was calculated to be the product of the test airfoil's drag and the vertical distance from the center of gravity to the drag's point of application. Knowing the test airfoil's moment contribution, an equation for the moment coefficient for the test airfoil was found. To get the equation for the $C_{m\alpha}$ for the test airfoil section, or $C_{m\alpha s}$, the derivative of the moment coefficient was taken with respect to the angle of attack. The vertical distance is the only term that was considered to vary as the angle of attack of the RPV increased. The drag was considered to act horizontally, independent of the angle of attack of the RPV. This assumption becomes more accurate the closer the test airfoil is to the nose of the RPV. Therefore, the equation for $C_{m\alpha s}$ equals a constant times the derivative of the vertical distance with respect to the angle of attack. The determination of a final equation for $C_{m\alpha s}$ is included below:



$$d = \sqrt{dx^2 + dy^2} \quad \text{where}$$

$$dx = X_{CG \text{ from L.E.}} + l_w - \frac{\bar{c}}{4} - l_s$$

$$dy = 0.833 \text{ ft.}$$

$$\theta = \tan^{-1}\left(\frac{dx}{dy}\right)$$

$$y_d = d \times \sin(\theta + \alpha)$$

$$C_{m_s} = \frac{M}{q_\infty S \bar{c}} = \frac{D_s(y_d)}{q_\infty S \bar{c}}$$

$$C_{m_{\alpha}} = \frac{dC_{m_s}}{d\alpha} = \frac{D_s\left(\frac{dy_d}{d\alpha}\right)}{q_\infty S \bar{c}} = \frac{D_s(dx \cos\alpha - dy \sin\alpha)}{q_\infty S \bar{c}}$$

$$C_{m_{\alpha}} = \frac{D_s\left(X_{CG \text{ from L.E.}} + l_w - l_s - \frac{\bar{c}}{4}\right)}{q_\infty S \bar{c}}$$

The second problem with the $C_{m_{\alpha}}$ equation for the RPV is that it assumed the distance from the center of gravity to the horizontal tail's quarter chord point, l_t , to be constant. This assumption is fairly valid for most aircraft, but in this study the center of gravity was a quantity whose range of variation was taken into consideration. The length, l_t , was therefore expressed as a function of the center of gravity as shown here:

$$l_t = 6.175 \text{ (ft.)} - X_{CG \text{ from L.E.}} - l_w + \frac{\bar{c}}{4}$$

Once these two adjustments are made, the resulting equation for the neutral point should be more accurate than that provided in the target analysis. The final equation for $C_{m_{\alpha}}$ is expressed here:

$$C_{m_{\alpha}} = C_{l_{\alpha}}\left(\frac{X_{CG}}{\bar{c}} - \frac{X_{ac}}{\bar{c}}\right) + C_{m_{\alpha}} - \eta \frac{S_t(6.55 - X_{CG \text{ from L.E.}} - l_w)}{S \bar{c}} C_{l_{\alpha}}\left(1 - \frac{d\epsilon}{d\alpha}\right) + \frac{D_s}{q_\infty S \bar{c}}\left(X_{CG \text{ from L.E.}} + l_w - l_s - \frac{\bar{c}}{4}\right)$$

The neutral point was then solved from this equation by setting $C_{m_{\alpha}}$ equal to zero and solving for the center of gravity. The neutral point equals the expression found for the center of gravity when $C_{m_{\alpha}}$ equals zero. The final equation found for the neutral point is as described here:

$$\frac{X_{NP}}{\bar{c}} = \frac{\frac{X_{ac}}{\bar{c}}(C_{L_{\alpha_w}}) - C_{m_{\alpha_i}} + A - B}{C}$$

where

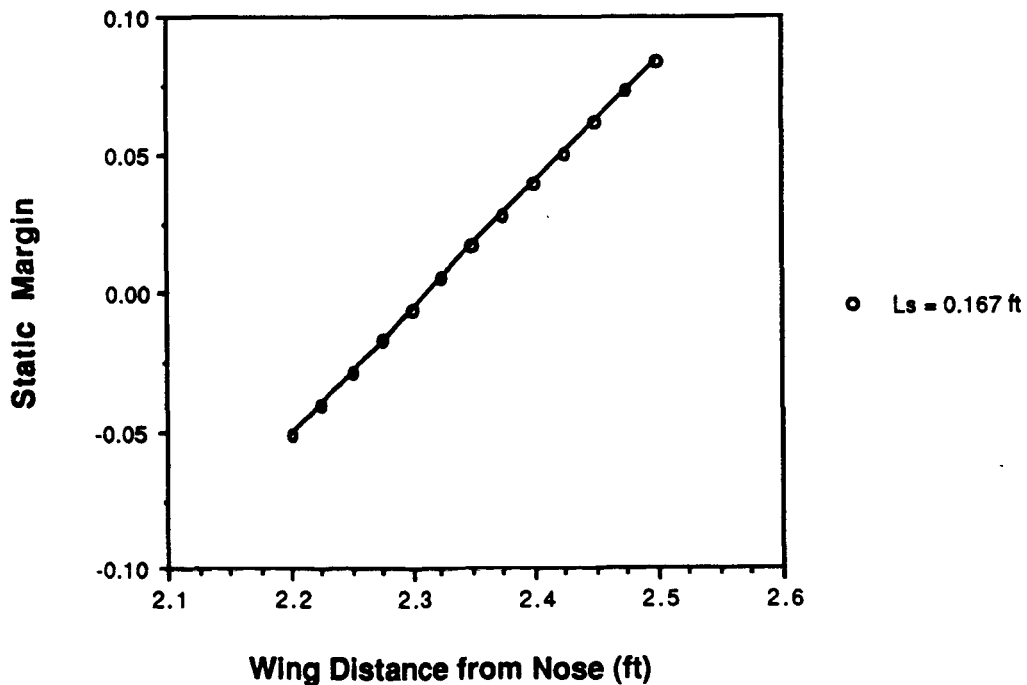
$$A = \eta \frac{S_i}{S \bar{c}} C_{l_{\alpha}} \left(1 - \frac{d\epsilon}{d\alpha}\right) (6.55 - l_w)$$

$$B = \frac{D_s}{q_{\infty} S \bar{c}} (l_w - l_s - \frac{\bar{c}}{4})$$

$$C = C_{l_{\alpha_w}} + \eta \frac{S_i}{S} C_{l_{\alpha}} \left(1 - \frac{d\epsilon}{d\alpha}\right) + \frac{D_s}{q_{\infty} S}$$

The static margin was found by taking the difference between the neutral point and the center of gravity. The following graph plots static margin as a function of wing location for the test airfoil mounted at the front of the fuselage.

Static Margin for Varying Wing Position with Fixed Test Airfoil



For this placement of the test airfoil, the quarter chord point of the wing should be located 2.425 ft. from the nose of the RPV to provide a static margin of 0.05. A value for the static margin of 0.05 is considered to be an acceptable value for most aircraft.¹

The neutral point location in the target analysis was 0.5 times the wing mean chord, versus 0.567 under the actual data analysis. The center of gravity location is 0.3 times the wing mean chord versus 0.517. Finally, the resulting static margin is 0.2 versus 0.05. This means that the center of gravity should be moved forward so that the actual data analysis will more closely match the target analysis. Another consideration is that the center of gravity should be moved closer to the quarter chord point of the wing than 0.517. In summary, the center of gravity should be moved forward in future design iterations of the Air Rhino, both to match control sizing more accurately and to lessen aerodynamic moments caused by the wing by matching the center of gravity and aerodynamic center.

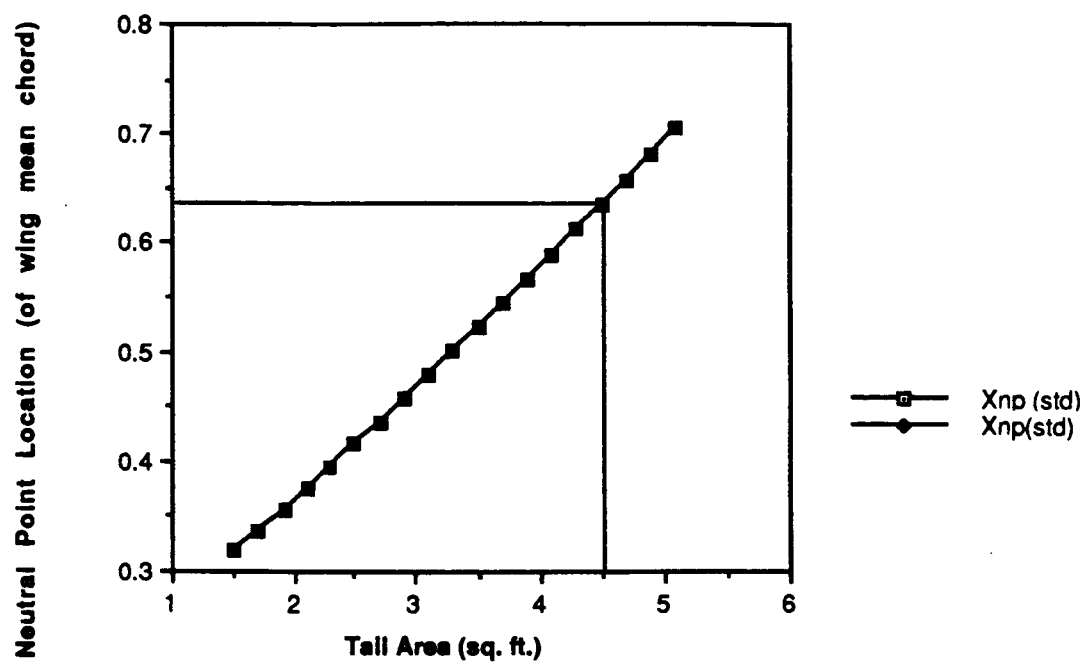
Surface Location and Sizing

The main parameters that affect the location of the stick fixed neutral point are the horizontal tail surface area, the horizontal tail aspect ratio, the tail length and the wing aspect ratio. The wing size and aspect ratio were primarily decided by the weight of the payload and the desired wing loading for the specific mission.

Now the tail surface area's affect on neutral point location will be examined further. A recommended first guess of the tail area being 20% of the wing area was made.² Below is a plot of tail area versus neutral point location. The distance from the center of gravity to the tail aerodynamic center (L_t) was fixed at 3.75 ft for calculation purposes. The tail area to maintain this length, then, was chosen to be 3.35 ft² which is about 23.5% the wing area.

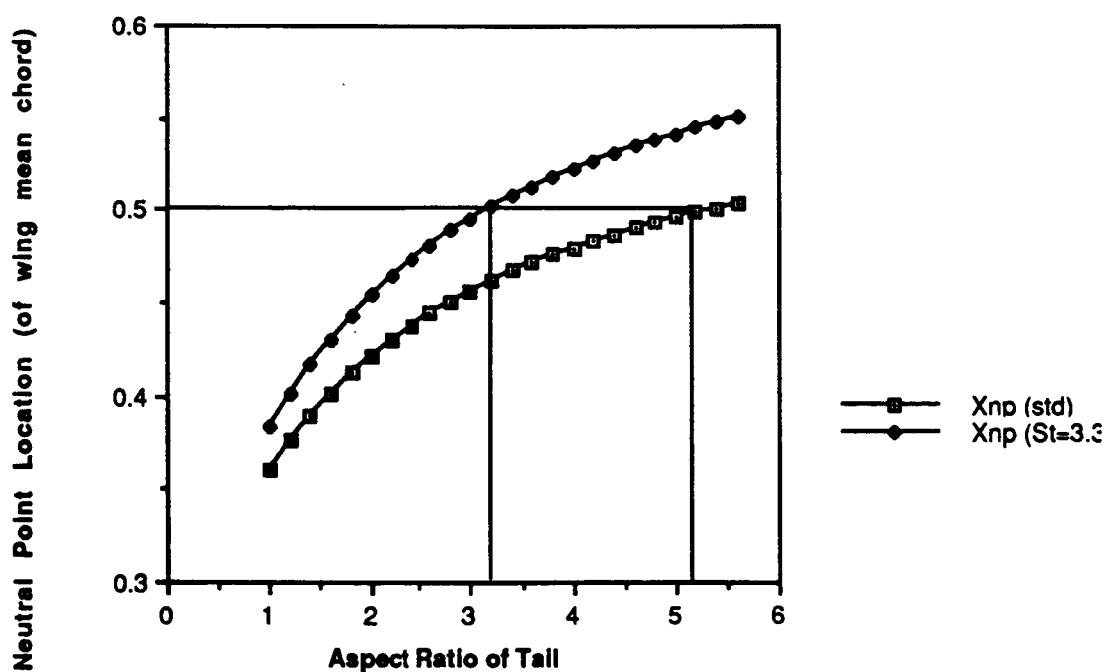
¹Nelson, Robert C., Flight Stability and Automatic Control, McGraw-Hill, New York, 1989, p. 64.

²Heinemann, Rausa, and Every, Aircraft Design.



Tail Area Effect on Neutral Point

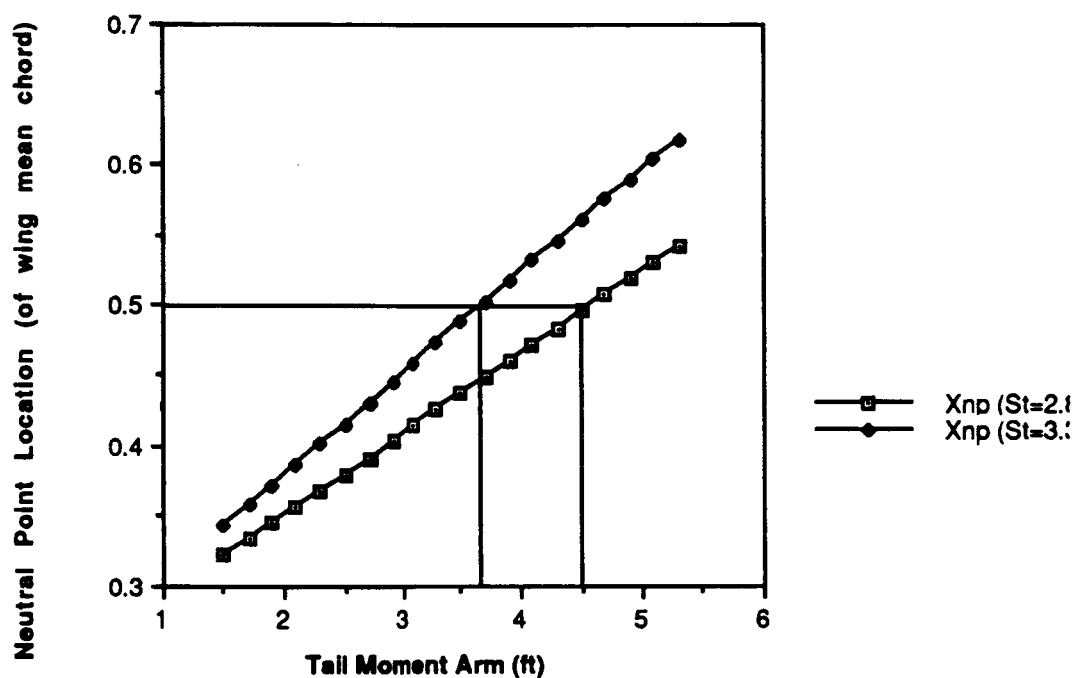
A plot of the neutral point location versus tail aspect ratio is below:



Tail Aspect Ratio Effect on Neutral Point

The graph shows a tail area of 2.85 ft^2 as well as 3.35 ft^2 . The aircraft's neutral point location moves rearward the higher the aspect ratio of the tail plane. The first guess for tail area of 2.85 ft^2 will achieve the desired neutral point location at an aspect ratio of 5.3. This implies that the span of the tail plane would be 3.88 ft and the chord would be 0.733 ft. But, with the tail area of 3.35 ft^2 , the desired neutral point of 0.5 times the chord of the wing is achieved at an aspect ratio of 3.35. This results in the tail span being 3.35 ft and a tail chord of 1.0 ft.

The next plot shows the neutral point location as a function of the tail moment arm:

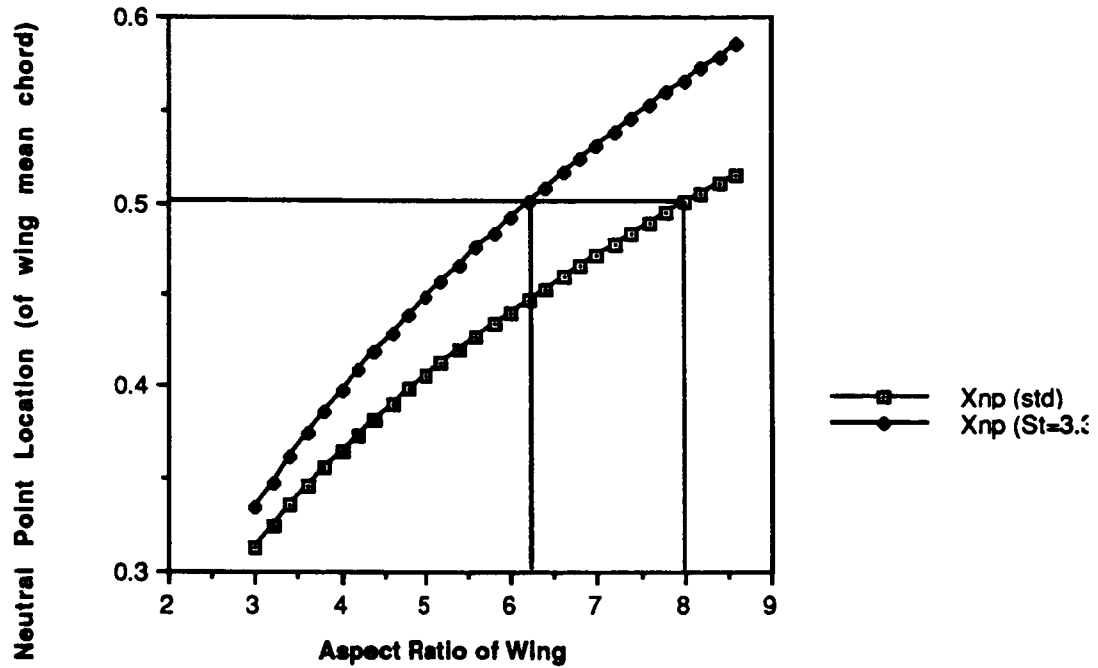


Tail Moment Arm Effect on Neutral Point

This is the distance from the center of gravity location to the tail's aerodynamic center. This plot has two curves for a tail area of 2.85 ft^2 (20% the wing area) and for a tail area of 3.35 ft^2 (23.5% of the wing area). The plot shows that to yield a neutral point location at 0.5 times the mean chord of the wing, the 20% tail area versus wing area would require a tail moment arm of approximately 4.5 ft. But, when the tail area is increased to

.235 times the wing area, the tail moment arm of 3.75 is obtained as was used for the previous analysis. Therefore, when these three figures are taken into account, a tail area of 3.35 ft^2 , a tail aspect ratio of 3.35, and a tail moment arm of 3.75 ft produce the desired neutral point location.

A final plot shows the neutral point location as a function of wing aspect ratio to make sure the aspect ratio of the wing was a good choice



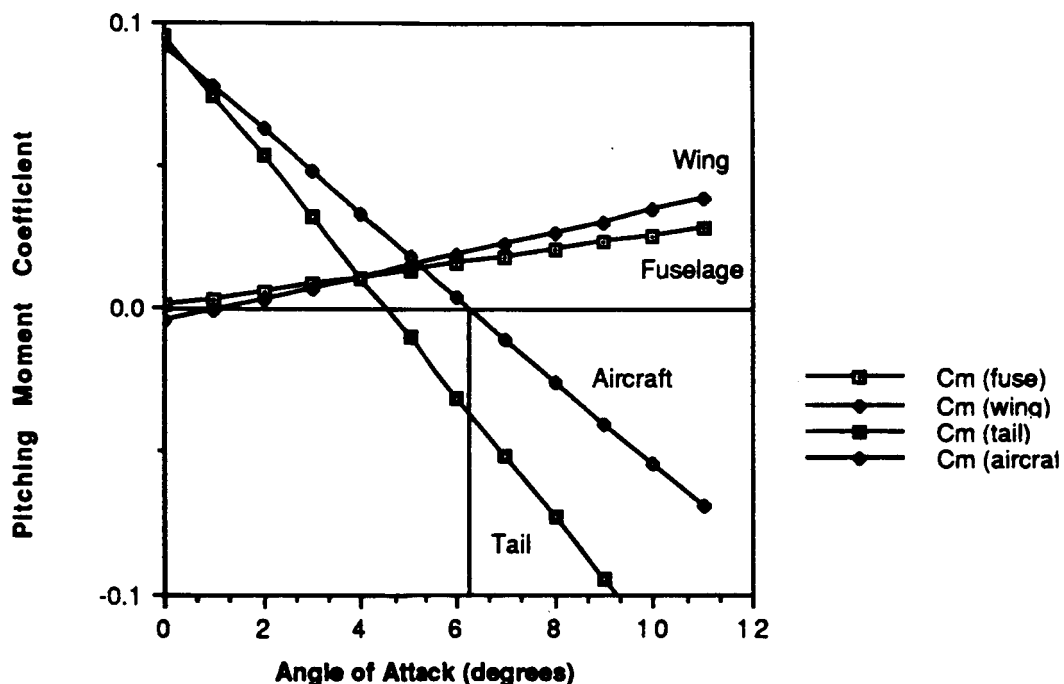
Wing Aspect Ratio Affect on Neutral Point

This plot shows that the neutral point travels further aft as the wing aspect ratio increases. With a tail area estimate of the tail area being 0.2 times the wing area and a tail moment arm of 3.75 ft, the wing's aspect ratio would have to be 8 to achieve the desired neutral point location. But, when the tail area is 23.5% of the wing area, the chosen aspect ratio of 6.33 is obtained.

Overall, then, the horizontal surface has been sized to produce the desired neutral point location for static stability. The horizontal tail then can be summed up as the following:

HORIZONTAL TAIL SIZING	
Horizontal Tail Area (S_h):	3.35 ft ²
Horizontal Tail Aspect Ratio:	3.35
Horizontal Tail Span:	3.35 ft.
Horizontal Tail Chord:	1.0 ft.
Tail Moment Arm (L_t):	3.75 ft.
Wing Aspect Ratio (AR)	6.33

Next is a plot of the pitching moment coefficient versus angle of attack for the RPV:



Pitching Moment Coefficient versus Angle of Attack

These results were obtained with a computer program that splits up the pitching moment into contributions from the fuselage, wing, and tail and provides the overall pitching moment coefficient for the entire aircraft. This plot shows that the RPV would cruise at an angle of attack of approximately 6 degrees when trimmed. The 6 degree fuselage angle of

attack was arrived at assuming a zero angle of attack for the wing and stabilizer with respect to the fuselage reference line. Also, it can be seen that the curve's slope is negative, as required for static stability.

This RPV, though, is designed for high speed flight, and a 6 degree cruise angle would increase the drag of the plane, an unwanted affect. But, as mentioned in the Wing Design section, the wing will be placed at an angle of one degree, reducing the angle of cruise to five degrees. This will reduce the fuselage cross section exposed to the free stream as the plane performs at high speeds. Still, this is quite a large angle of attack and some compromise should be considered to further increase the wing incidence angle to reduce the cruise angle of attack. The aircraft should be as trim as possible since it will be flying at such high speeds. A wing incidence angle of around three degrees would be recommended.

Utilizing the same program as above and with a tail incidence angle of zero degrees, the elevator trim angle was found to be -0.4455 degrees. Such an upward deflection would produce negative lift which is in agreement with the function of the horizontal tail. Because the design of this specific aircraft is such that it utilizes an all moving stabilator, the stabilizer angle of incidence parameter may be adjusted such that the final elevator trim angle would be zero.

Finally, the same program was run with the preceding parameters to obtain a neutral point location of 0.489 times the mean wing chord length. This is only a 2% difference from the target analysis solution of 0.5 times the mean wing chord length. So, the above parameters are correct as they stand.

The next surface that has to be sized is the vertical tail area. This aircraft has two vertical tails since the plane has a twin boom configuration. The vertical stabilizers in this aircraft must not only provide lateral static stability but must also provide sufficient control power to overcome both the adverse yaw produced by the deflected ailerons and the lift force of the test airfoil.

The vertical tail area was initially chosen to be 0.14 times the surface area of the wing, a typical RPV value. This led to an area of approximately 2.0 ft². Twin vertical tails imply an area of 1.0 ft² per vertical stabilizer. Since the chord of the horizontal tail stabilizer is 1.0 ft, it would be convenient, from a construction point of view, to make the chord of the

vertical stabilizers 1.0 ft also. This would mean a vertical height of 1.0 ft. However, from an aerodynamic aspect, it is always advantageous to have an airfoil that has an aspect ratio greater than 1. The only other way of maintaining the same 1.0 ft base chord while having a greater than one foot vertical height would be to have a tapered surface. In this instance, aerodynamic characteristics may be sacrificed for ease of construction.

Vertical tail volume ratio gives a more thorough analysis of lateral static stability because it takes into account the tail moment arm. But the tail moment arm was set at 3.75 ft by the longitudinal static stability requirements, as seen above. The vertical tail volume ratio, using a 1 ft chord and a 1 ft vertical height for each tail, produced a vertical tail volume ratio of 0.35 which is over 100% larger than typical values for RPVs. However, some extra control power is needed to counteract the test airfoil, so these values were left intact.

The results of the vertical tail sizing are summarized below:

VERTICAL TAIL SIZING	
Vertical Tail Area (S_V):	2 @ 1 ft ²
Vertical Tail Aspect Ratio:	1
Vertical Tail Span:	1 ft.
Vertical Tail Chord:	1 ft.
Tail Moment Arm (L_t):	3.75 ft.

Dynamic Stability

In order to make the RPV easier to fly, an on board automatic piloting system will be used in conjunction with a feedback control loop linked to the data acquisition package.

One recommended automatic pilot is the B.T.A. Automatic Piloting System built by Cal Orr Custom Electronics.¹ This automatic pilot is a two degree system. For pitch, the elevator servo is controlled and for roll, the aileron servo. Basically, the system maintains airplane position according to the position of the control sticks. Therefore, if the sticks have been calibrated to have level flight at the center position, the control system will

¹"B.T.A. Automatic Piloting System", in Scale RC Modeler Volume 15, Number 2, February 1989, pg 14-16.

maintain that attitude indefinitely. It will compensate for wind gusts and other outside interferences. When the sticks are changed to perform a turn, the control system will ensure a level turn without the loss of altitude.

Landing will also be easier since the automatic pilot will keep the plane at the desired attitude even when wind gusts and ground effects develop. This automatic pilot features an on/off switch which allows the system to be turned on in flight. When turned on and with the sticks pulled totally back for a climb the RPV will climb at a maximum climb rate. If the Air Rhino unexpectedly flies out of line of sight, a preprogrammed control input may be used to turn the plane around before it travels out of radio range. Finally, the automatic pilot can right the plane if the system is turned on while the Air Rhino is upside down. Overall, the automatic pilot takes a lot of work away from the pilot. Therefore, he can spend more time making experimental adjustments and less time just keeping the plane level.

Another safety feature could also be designed in with the use of this automatic pilot. Since the system can be turned on or off during flight, the off position could be calibrated so that the plane flies in a circle. Therefore, if something goes wrong with the RPV, the automatic pilot could be turned off and the RPV would loiter in a circle until the problem could be solved. This would be a good thing to have for a safety precaution.

A final good point about the automatic pilot is its size, weight, and cost. The dimensions of the control box are 6.5 in X 2.2 in X 2.0 in, which allows the system to fit in the Air Rhino. It weighs, though, only about 1.0 lb while drawing 150 ma of current. The receiver and automatic pilot will have its own battery pack separate from the data acquisition system. So the plane will still be controllable even if the central processing unit and associated systems fail. Finally, the cost of this system is only \$600, a reasonable sum for such a useful addition.

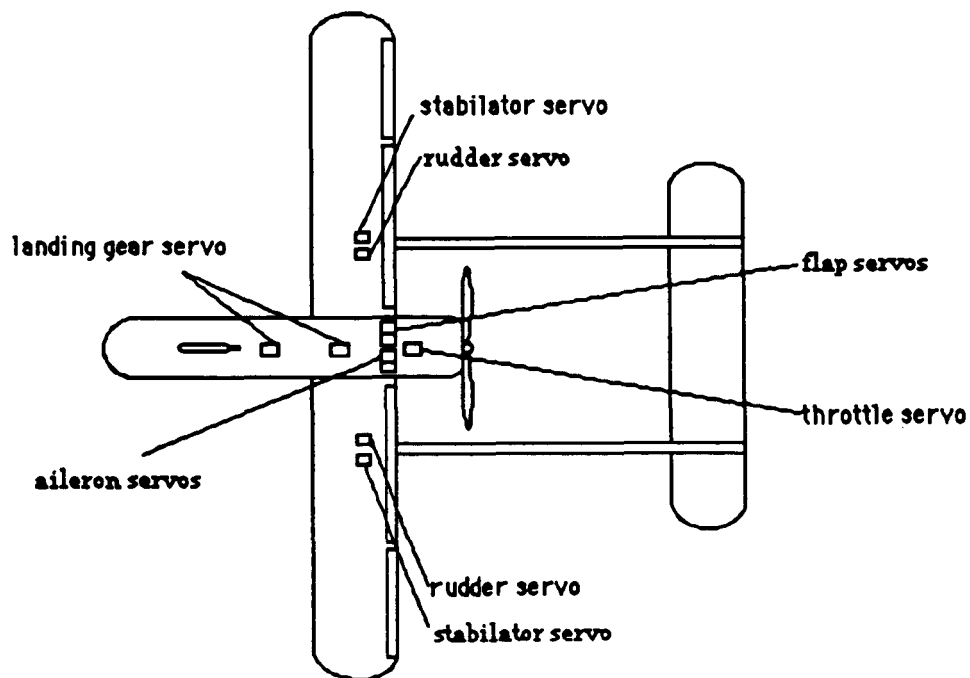
An automatic pilot is almost a necessity for the Gold Mission in particular. Because the test airfoil will be rotated through different angles of attack during flight, dynamic control during test section rotation would be difficult without a stability augmentation system. The automatic pilot will ensure level flight as the test airfoil changes position. The plane will neither roll nor pitch due to aileron and elevator control. As for yaw control, a control loop will be set up with the data acquisition computer and

with the rudder servo.

Once the test airfoil is rotated by a step motor to a specific setting, the feedback control loop between the rudder servo and the data acquisition system will maintain the angle of attack of the test airfoil while the data is collected. Therefore, even though the test airfoil deflections are destabilizing, the automatic pilot and computer control of the rudder will maintain level flight. All of this, then, will provide the best possible conditions for data acquisition in an otherwise unsteady environment.

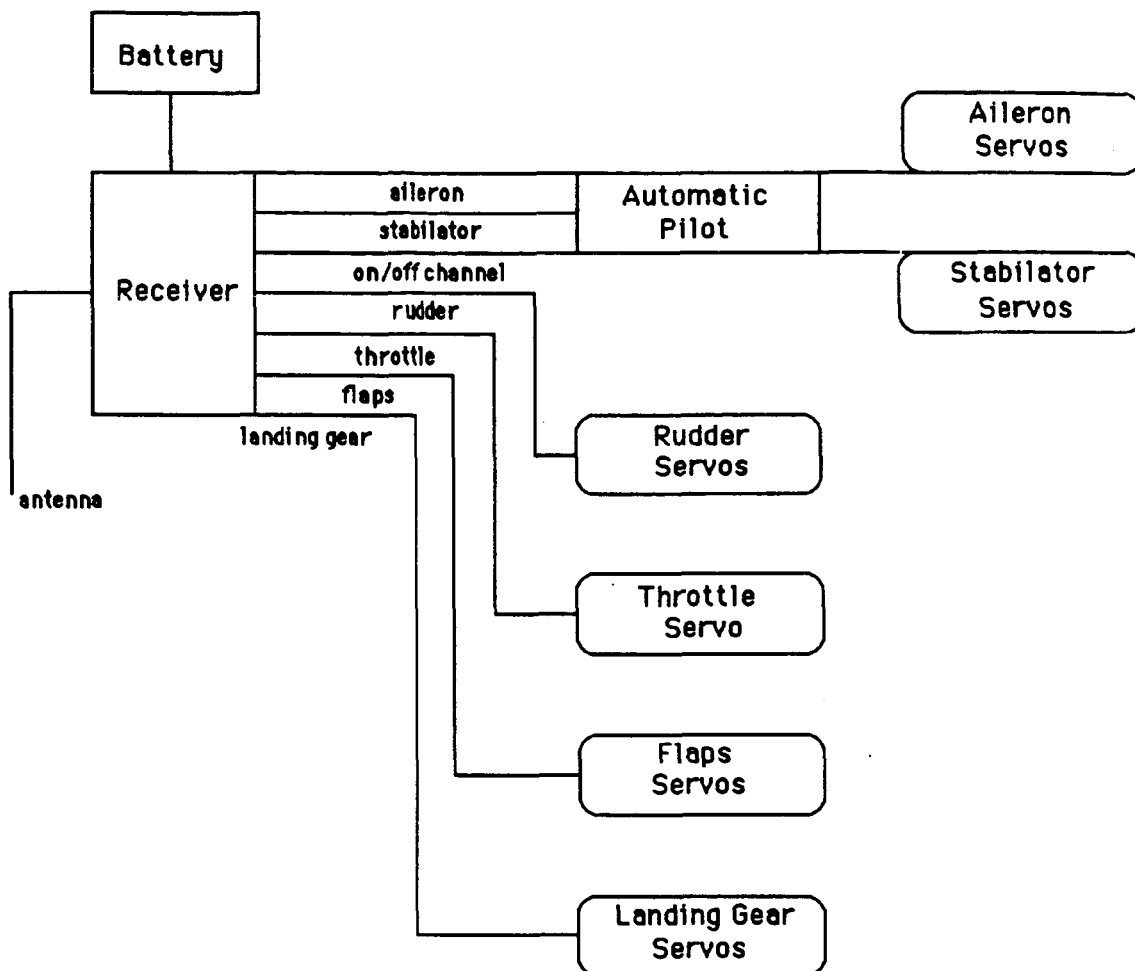
Control Mechanisms

As stated, the Air Rhino will be utilizing ailerons for roll control, rudders for yaw control, and full stabilators for pitch control. Six servos will be needed for these control surfaces. A seventh servo will be used in conjunction with the throttle. Two more will be used to control the flaps. Finally, two more servos will be used to deploy and retract the landing gear. Overall, then, eleven servos will be used to control the various mechanisms of this RPV. The various positions of these servos are shown below:



Servo Location

As a result of the need for so many control actuations, a seven channel receiver will be used. This system is pictured below:



(The mechanisms that employ two servos each like flaps will all plug into adapters and then plug into only one channel). The various servos and the auxiliary on/off channel will all be connected to this receiver. The transmitter will have levers to control the ailerons, rudder, stabilator, and throttle, as well as a landing gear switch, flap switch, and autopilot on/off switch.

The servos for the stabilator will be located in the right and left wings of the plane in front of the tail booms. Control push/pull cables will then link directly back through the booms and connect to the stabilators. The

rudder servos will share the same position in the wings but in a location slightly closer to the fuselage. Control cables will also be connected to the rudder through the tail booms. Wire leads from the receiver to the servos in the wings will be insulated so that no electrical noise will influence the servos' behavior. Also, hatches will be built directly above the servos in the wings so that they can be accessed readily for repair or replacement.

The rest of the servos will be housed in the fuselage itself. The landing gear servos will be located directly over the landing gear hinges, and the gear will fold inward. The flap and aileron servos will be located in line with the back portion of the wings so that direct linkages can be trailed along the span of the wings. Finally, the throttle servo will be placed in proximity to the throttle of the motor itself. Overall, this design limits the complexity as well as the length of the control rods in order to make the Air Rhino simpler to manufacture.

One important control problem with the rudder/aileron/stabilator setup is the moments caused when the test airfoil is deflected to high angles of attack. A simple free-body diagram will show that the rudders alone cannot counteract both the moment and the force applied by the test airfoil. The large vertical tail surfaces will help some, but additional vertical surface may be required to prevent excessive "crabbing" or lateral translation of the aircraft. The addition and sizing of these units would require a full-blown dynamic stability analysis of the Air Rhino, but design has not yet progressed to that point.

The Flaps System

Since the Air Rhino is being designed to reach a maximum speed of 200 ft/sec, weight and engine considerations set the stall velocity at 54 ft/sec. This produces an acceptable takeoff distance (with a 50 ft obstacle height) of 275.9 ft, but the landing distance is outside the limit of 300 ft, at 710 ft. This distance can be chopped considerably by reducing the stall velocity, and thus the landing velocity. Also, a smaller stall velocity is desired to allow data acquisition tests over a greater range of Reynolds numbers. Since this is one of the design goals, anything that would reduce the stall velocity would be beneficial.

In order to decrease the stall speed as well as the landing and take off distances, flaps have been added to the Air Rhino. This is an efficient way

to make this plane more airworthy. Since it is relatively simple and easy to build, the split flap was chosen for this specific mission. The split flap increases the lift while adding to the pressure drag due to the wake produced by its deflection. The split flap does produce a small nose down moment to the wing, but the relatively large horizontal stabilizer will help counteract this moment. Also, the split flap almost maintains the maximum angle of attack of the wing while deployed. Overall, the split flap is easy to build and is easy to integrate into the RPV. Deflections are easy to perform by the servo since a simple rotation is all that is needed. The split flap does not slide back as it is being deployed.

The dimensions of the flaps were found using graphs taken from Stinton.¹ The flap chord was chosen to be 20% of the wing chord since this does not add to the profile drag of the wing at all. The maximum deflection of the flaps should be kept under 40 degrees to keep the drag produced by the wing down. Finally, the flap span was chosen to be 60% of the wing span in order to have its effects felt across most of the wing. The remaining part of the wing is taken up by the ailerons. Overall, the use of this split flap system adds a 0.5 increment to the lift coefficient of the wing, and at maximum deflection produces a profile drag increment of 0.06.

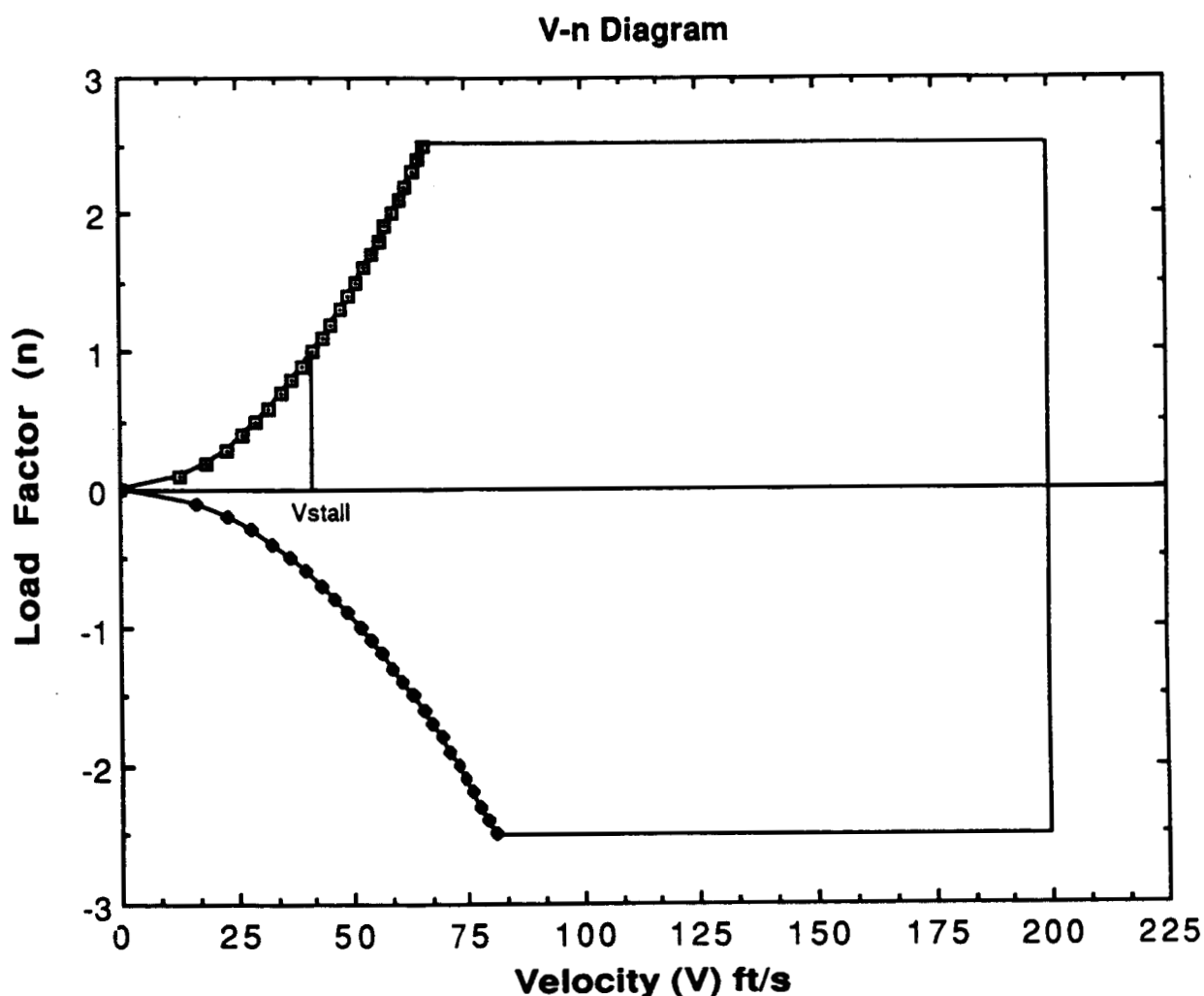
When these flaps are deployed, the minimum velocity is reduced by 23.2% to 41.5 ft/sec. The take off distance changes to 205 ft. This is a 25.7% decrease, quite good for adding only split flaps. The landing distance, though, is reduced by 52.0% to 341 ft. This decreases because both the added lift and added drag affect the landing distance in a positive manner to shorten it. Overall, the landing distance still does not meet the design parameters, but the addition of flaps cuts it drastically down.

¹Stinton, Darrol The Design of the Aeroplane, Von Nostrand Reinhold Company, New York (1983).

Structural Design

V-N Diagram

The velocity versus load factor (V-n diagram) can be seen below. The main features of this plot are the maximum and minimum load factors of 2.5 and -2.5, respectively. The maximum speed of the RPV is 200 ft/s while the stall speed is 41.5 ft/s. The wind gusts of 10 ft/s do not affect this diagram because of the compensating effects of the autopilot systems. Maximum positive lift coefficient with flaps is 1.22 and maximum negative lift coefficient is -0.8.



Flight and Ground Loads

The flight loads on the Air Rhino are dependent on the load factor expected and the weight of the plane. As can be seen in the V-n diagram above, load factors above $n=2.5$ are not expected to be encountered by the Air Rhino during flight. The effects of gusts can be neglected since the autopilot system will compensate for them.

Because the materials chosen for the wing are dependent somewhat on the amount of money available to spend, a detailed analysis of forces within the wing has not been conducted. Some sort of stress and strain analysis would have to be conducted before actual production of the Air Rhino. However, a rough estimate of maximum wing stress can be made using the following equations for load factor and inertial force:

$$\begin{aligned}
 n &= \pm 2.5 = 1 + a/g \rightarrow a_{\max} = -3.5g \\
 F_i &= ma = \frac{-W_{\text{wing}} \cdot 3.5g}{g} = -5.4 \text{ lb} \cdot -3.5 = 18.9 \text{ lb} \\
 F_{\text{lift}} &= C_l \cdot r \cdot V^2 \cdot S = 1.12 \cdot 0.002377 \cdot (200)^2 \cdot 15 = 160 \text{ lb} \\
 F_{\text{tot}} &= F_{\text{lift}} + F_i = 178.9 \text{ lb}
 \end{aligned}$$

If this load is split between the two wings, there will be a maximum force of 89.5 lb on each wing.

Ground loads will be transmitted through the three landing wheels to the airframe. As the Air Rhino lands, a flare maneuver by the pilot shortly before landing should give a landing glide angle of about 5° . Given a tire and gear give of 1.0 in, and no bounce on landing, the following derivation yields the inertial force on the plane:

$$\begin{aligned}
 D/R &= \sin q \cdot 1.2V_{\text{stall}} = \sin 5^\circ \cdot 1.2 \cdot (41.5 \frac{\text{ft}}{\text{sec}}) = 4.34 \frac{\text{ft}}{\text{sec}} \\
 t &= \frac{\text{tire "give"}}{D/R} = \frac{0.8333 \text{ ft}}{4.34 \text{ ft/sec}} = 0.0192 \text{ sec} \\
 a &= \frac{V_{\text{bar}}}{t} = \frac{(4.34-0) \text{ ft/sec}}{2 \cdot 0.0192 \text{ sec}} = 113.01 \frac{\text{ft}}{\text{sec}^2} \\
 F_i &= ma = \frac{35 \text{ lbf}}{32.2 \text{ ft/sec}^2} \cdot 113.01 \frac{\text{ft}}{\text{sec}^2} = 122.8 \text{ lbf}
 \end{aligned}$$

Once this inertial force has been determined, a free-body diagram can be constructed to find the forces on each gear assembly due to the

inertial force and the plane weight. Note the assumption that the main gear is placed at 0.9 chord lengths behind the leading edge of the wing, and the nose gear is one foot behind the nose. Also assume the plane lands "on the level." The forces found thus are 32.7 lb for the nose gear and 62.5 lb for each of the main gear assemblies. Obviously some structure stiffening will have to be added to compensate for these loads.

Materials Selection

The twin boom structure is one critical component of the airplane design. The proper construction and design of the twin booms is important for the structural integrity and control characteristics of the aircraft. Because they are the sole connection between the control surfaces and the rest of the plane, they must be able to transmit all control forces from the control surfaces to the fuselage and wings.

The length and complexity of applied forces makes the twin booms very vulnerable to bending, twisting, and snapping. A rigid twin boom structure is needed to help prevent dynamic instability and transmit control forces efficiently. Also, tight movement of the control surfaces with little play is dependent on a rigid boom structure, since control cables must run from the controls through the booms to the centrally-located servomotors.

A worst case scenario was established to set the strength criterion for the twin booms. The worst case condition for the maximum lift force on the tail section was determined to be when the horizontal stabilizer was under full elevator deflection at maximum speed. Based on a maximum lift coefficient of .9 for the horizontal tail surface, a lift coefficient is increased by 47% under full elevator deflection, and a maximum speed of 200 ft/sec, the boom is subjected to a 200 lb force due to the tail section.

One very important aspect of the booms is their shape. Several different cross sectional designs were examined to determine which would perform best. The booms were modeled as ideal beams under pure bending and examined under identical loading conditions. In order to determine which design was best, each geometry was modeled with the same cross sectional area and material thickness. The geometry which yielded the lowest maximum stress in its members would be the ideal geometry. The material in each beam design was modeled to be 10 in. long and 0.25 in. thick, with 2 in.² total cross sectional area. A point load of 100 lb., located at

the tip of the beam, was used to model a normative axial stress. The beam designs were also checked for shear stress using a 100 lbf.-in. torsional load. The different designs and the stress calculations associated with each are provided below:

RESULTS OF STRESS TESTS OF DIFFERENT GEOMETRIES		
geometry	max axial(psi)	max torsional (psi)
circular, 1.4 in. outside radius	852	38.8
rectangular, 2 in. x 2.5 in.	779	40
I-beam, spar 3 in., web 2.5 in.	543	2623
triangular, 2.67 in. edge	2170	32.6

Based on these results, the box structure and hollow tube design proved to be best. However, properties of the material chosen will interact with geometric properties to ultimately determine which design will be the best for construction.

Structural materials were chosen based on several critical material characteristics. These characteristics, in order of importance, are strength, flexibility, weight, and cost. The different classes of materials examined were polymers, ceramics, metals, composites, and woods. The table below shows the relative rankings of these materials over the critical material characteristics:

RELATIVE RANKING OF MATERIALS CLASSES					
Material Class	Strength	Cost	Weight	Machinability	Availability
Wood	4	1	2	1	1
Metal	2	2	4	2	2
Composites	1	4	1	3	3
Ceramics	3	3	3	FAIL	4
Polymers	FAIL				

To obtain the best material, each was evaluated to see if they were up to standard in each of the critical characteristics. All of the materials except the polymers were able to sustain the maximum stress realized in the box construction under the worst case loads. Ceramics were then

eliminated because of their lack of ductility and machinability. This left metals, woods, and composites to be examined for their weights. A comparison of typical densities for these classes appears below:

DENSITIES OF VARIOUS MATERIAL CLASSES					
Metals		Woods		Composites	
Steel	.284	Balsa	.005	Fiberglass	.005
Titanium	.162	Spruce	.013	Plywood	.025
Aluminum	.100	Douglas Fir	.017		
Tin	.264	Pine	.024		
Halstoy	.297				

In comparison, the lightest metals weigh over 4 times as much as the heaviest woods, while the composites weigh about 1/3 of the lightest wood. Thus, metals were eliminated. Finally, woods and composites were examined for cost. If cost were a minor concern, it would be highly advantageous to use the more expensive composites in construction, shaping them into circular tubing. However, since cost is a significant concern in the proposed aircraft, wood is the logical choice. Although wood has other positive aspects such as availability and machinability, it is important to determine which type of wood is best suit for the construction of the booms.

There are several possible types of wood to choose from. The most commonly used in RPV applications are spruce and balsa. Each of these meet the material selection criteria and are viable choices. Spruce is found to have an modulus of rupture of 11,200 psi while balsa's is 7400 psi. Even though balsa weighs about half as much as spruce it only provides about 2/3 of the strength. Since strength is of more importance than weight, spruce was chosen.

Materials choice for the airframe of the plane were made using similar criteria to those used for the tail boom. The airframe, which takes the loads from the landing gear, the twin booms, the force balance setup, and the wings, must be strong enough to carry any loads experienced under the chosen load factor limit of $n=2.5$ in the air, and impact loads during landing or crashes. Because of the landing loads, which are high compared to the normal wing loads, some metal stiffening may be necessary for the inner, non-dihedral wing.

The skin of the fuselage is not load-carrying, and so materials choice for the skin is much wider than for the airframe or the twin booms. Mylar "Monokote" surfacing was used on the technology demonstrator with good results.

One place where metallic structure may be required is the engine mounting. To avoid setting the plane on fire, some clearance must be allowed between the engine and any nonmetallic parts. Mounting the engine in a semi-enclosed position will reduce this problem somewhat as the engine will be cooled by convection to the freestream. Another option would be wood coated with epoxy or some other agent.

Takeoff and Landing

Takeoff and landing of the Air Rhino proved to be more difficult than first envisioned. The Gold Mission requirement of a 150 ft. radius circle for landing and takeoff with a 50 ft. object clearance was difficult to meet given the plane weight and airfoil choice. Also, because fragile electronics and strain linkages will make up much of the cargo of the Air Rhino, a soft landing is critical to reduce fatigue and improve overall system life span. Landing systems were considered more critical than takeoff systems because of this criteria.

Several landing systems were considered, including gearless and conventional gear, a parachute landing system, a 'fly-into' net retrieval system on the ground, a gearless 'bellyflop' conventional landing, and a vertical-drop net retrieval system.¹ Gearless landing was ruled out by the pusher propeller configuration and its placement of the propeller dead center and low enough to touch the ground. The vertical-drop net was also eliminated, due to expected difficulty in piloting the Air Rhino with the necessary degree of precision.

The three remaining landing systems were judged on the following criteria:

- SAFETY. The plane should minimize both the risk of inadvertent damage to ground objects and the amount of damage sustained in case of accident.
- REUSABILITY. The landing system should minimize loads experienced by components of the plane so they will not break. In the Air Rhino, the force balance and telemetry systems are especially fragile and important to protect.
- TURNAROUND. The landing system should be easy to reload and/or reinstall for rapid turnaround time.
- LANDING AREA. The landing system should minimize the ground area needed for a successful landing and safe operation.
- WEIGHT, SIZE, COST. These quantities should all be minimized.

These categories were weighted in accordance with the group design

¹ University of Notre Dame Department of Aerospace Engineering, Launch and Retrieval Systems for Remotely Piloted Vehicles, Fall 1988.

goals discussed in 'Concept Selection Studies'.

There are a few basic parameters that have strong influence on the performance of landing systems. Since most landings are performed at as low a speed as possible, plane stall speed is obviously an important factor. Stall speed is determined by the equation $V = \sqrt{\frac{2W}{\rho \cdot C_{l_{\max}} \cdot S}}$. In this equation, density ρ is assumed constant over the flight environment (assumed sea level) and maximum lift coefficient $C_{l_{\max}}$ is dependent solely on airfoil choice ($C_{l_{\max}}=0.72$ for the Air Rhino). The weight W and the wing area S are important parameters in determining landing characteristics.

Since landing takes place vertically, the two main sources of vertical force on the plane -- lift and weight -- are also critical. Weight was already on our 'parameters list'. Lift at landing is dependent primarily on $C_{l_{\text{land}}}$ and landing speed. $C_{l_{\text{land}}}$ is again dependent largely on airfoil choice, and landing speed is dependent on the aforementioned stall speed. Once again, weight and wing area appear to be critical parameters.

Inertial forces on the plane at landing are dependent on the descent rate and the distance the plane has to decelerate. These two quantities are considered as parameters solely for the landing forces analysis.

Besides the restraints imposed by the pusher configuration of the airplane, there are also some quantitative constraints on landing system design. One is the amount of force that the force balance was capable of withstanding. Since the force balance must be calibrated to measure the loads on a typical test airfoil (estimated independently at 60 lb), loads on the force balance should not exceed this by much to avoid overstraining the connecting spike. The other constraints are listed in the 'Gold Mission Request for Proposals,' which states that "takeoff and landing must be accomplished in a circular area with no greater than a 150 ft radius," and that turnaround time must be no greater than 15 minutes.¹ See the Review of Design Requirements section for more details.

The three systems were ranked in six weighted categories on a 10 point scale. The best performer in a given category was assigned the maximum score of 10, and others were rated relative to the best performer.

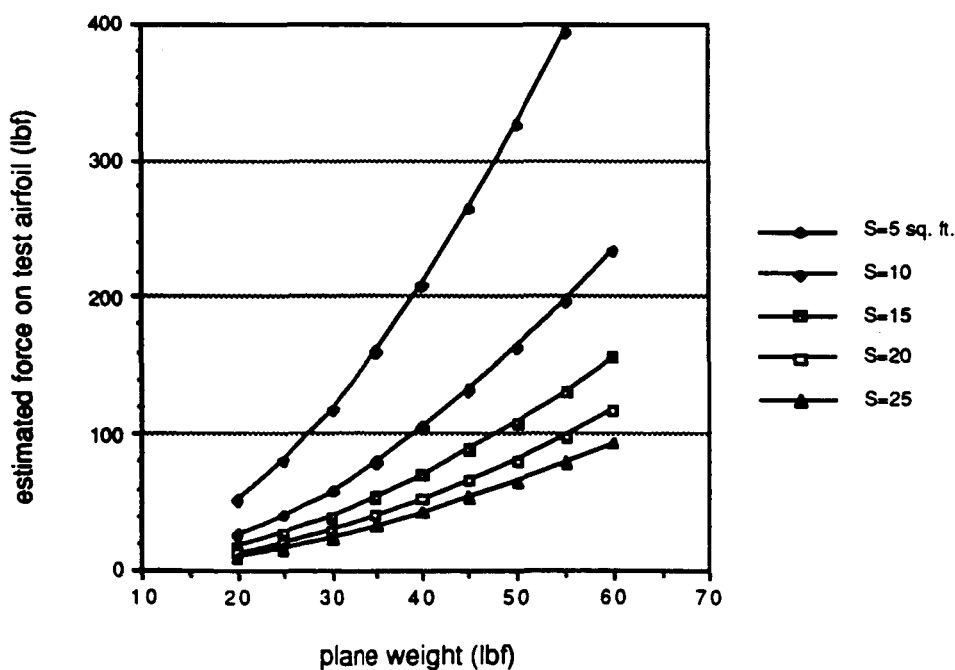
¹ University of Notre Dame Department of Aerospace Engineering, Request for Proposals, Spring 1989.

The results of the rankings are tabulated below:

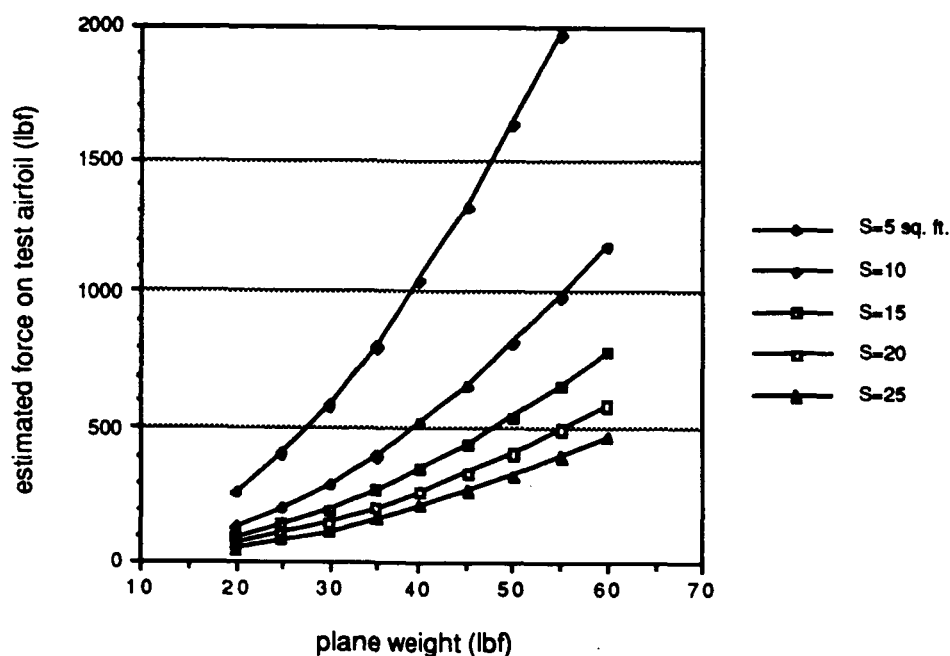
RESULTS OF RELATIVE RANKINGS							
category	weight	ranks			points		
		chute	gear	net	chute	gear	net
Plane Component Loads	30%	10	10	2	3.0	3.0	0.6
Safety	30%	9	10	6	2.7	3.0	2.1
Landing Area	8%	7	5	10	0.56	0.4	0.8
Weight and Size	8%	5	7	10	0.4	0.56	0.8
Cost	8%	9	10	8	0.72	0.8	0.64
Reload Time	16%	4	10	6	0.64	1.6	0.96
SUM OF RELATIVE RANK POINTS					8.02	9.36	5.90

From the results of the table, it is clear that the net system is not recommended for the Air Rhino. The net's main shortcoming was the external forces applied to the force balance during landing. The graphs below illustrate best and worst cases for forces experienced by the force balance during a net landing:

Forces on Test Airfoil During Net Landing (best case)



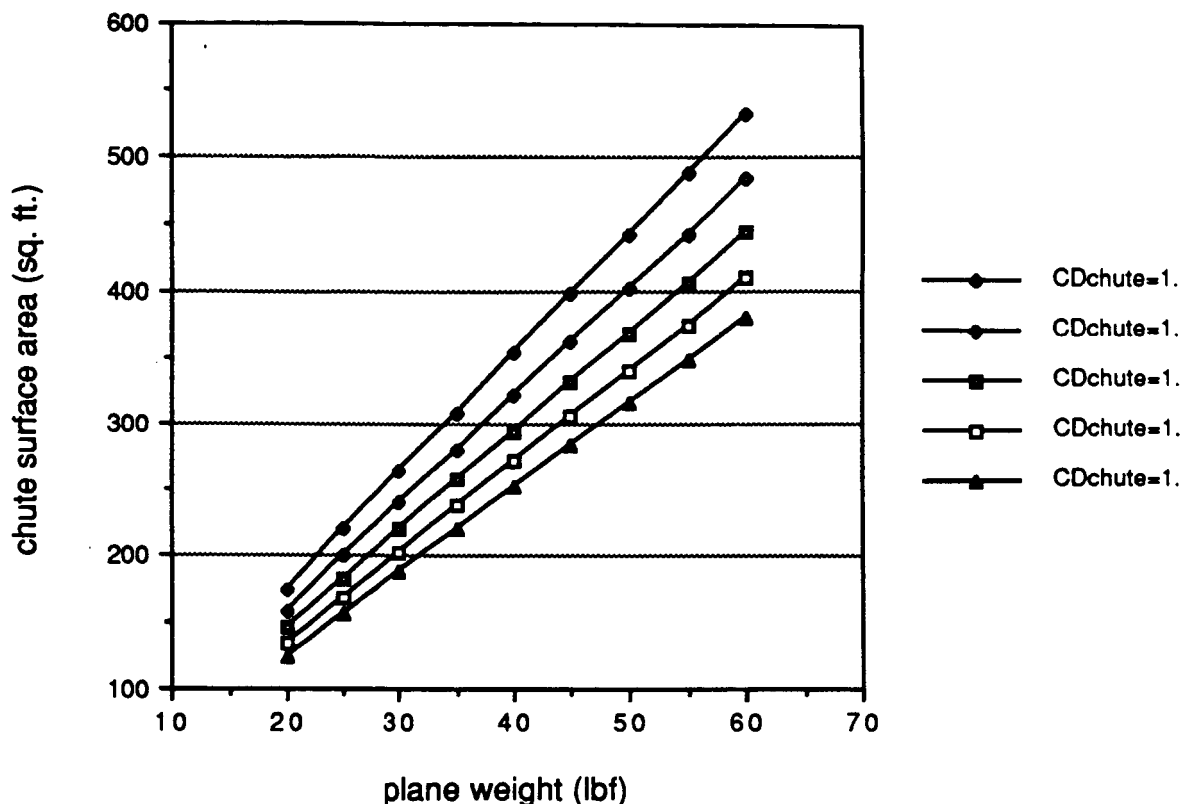
Forces on Test Airfoil During Net Landing (worst case)



The best case assumed a 5 ft deceleration distance and a load on the test airfoil totalling 10% of the total load on the plane. The worst case assumed a 2 ft deceleration and 20% force carried. The test airfoil forces, even in the best case, exceeded 50 lb, and in the worst case were as high as 270 lb. In contrast, since the gear and parachute systems transmit external forces solely through the gear or pads attached to the bottom of the plane, the only forces on the force balance are relatively small inertial forces, generally well under 20 lb.

The parachute system was eliminated primarily because of its high weight, size and cost. A graph showing the predicted chute size for chutes with various drag coefficients is shown below:

Parachute Sizing (R/D=10 f/s, S=15 sq. ft.)



The chute size required for a 10 ft/sec descent rate landing was 260 ft², which translates to a cubic volume of 374 cu in and a weight of almost three pounds, assuming a chute thickness of 0.01 in and density of 1 oz per square yard. Chutes for a 5 ft/sec descent rate were even larger. Add to this the chance of a bad chute packing and the expense of parachutes, and it seems the parachute is not the best choice.

The gear system has relatively few safety problems. The landing area is well defined. Consequences of component failure are not great; the plane would probably be damaged but damage to ground objects would probably occur only if the propeller were to be broken while spinning. Other pieces of debris would not travel far before they hit the ground. Landing speed is high, but the landing path is fairly easy to predict.

Conventional gear weight and size depends somewhat on the choice

of fixed vs. retractable gear. For low-speed RPVs, the fixed gear's drag penalty is not enough to justify the added complexity of retractable gear. However, the high target speed of the Air Rhino, on the order of 200 f/s, necessitates retractable gear; swinging the gear out of the way will reduce the gear drag to as little as 10% of the drag when extended.¹ The gear itself weighs about one pound; additional airframe stiffening and apparatus to rotate the gear up would weigh at most an additional pound. Size of the gear would be small compared to the size of the plane -- at most, 30 cu. in.

The gear system cost consists of materials cost for the tires, struts, retraction mechanism, and additional structural support for the gear. The tires and struts are relatively inexpensive, but the retraction mechanism is liable to be more expensive and the structural alterations, while difficult to predict, will incur some additional cost.

The gear system does not require any reloading, other than a quick check of the retraction mechanism. Recovery is also simplified by a fairly predictable landing pattern. Testing of the landing gear system after landing should take two minutes at most.

Since the landing system chosen was conventional gear, it only makes sense to use the same gear for the launch system as well. Adding some sort of elastic assist to the plane for takeoff was contemplated, but data on elastic properties of suitable bungi cords was difficult to find. Elastic assist is one area that could merit further study.

¹Sighard F. Hoerner, Fluid-Dynamic Drag, published by author, Midland Park NJ, Chapter 13.4

Part III. Performance

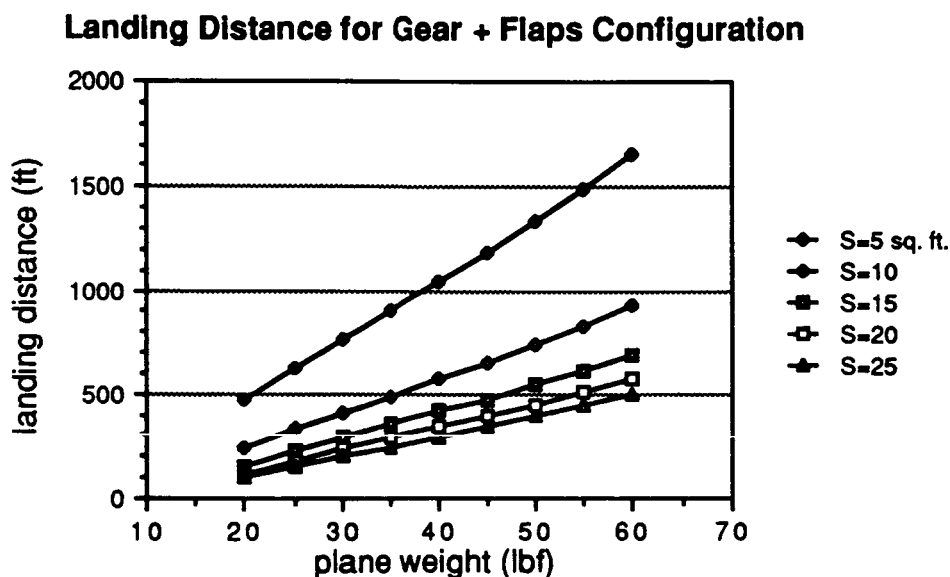
III. Performance of the Proposed Plane

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Takeoff and Landing Distances

Although landing gear clearly is the best choice for the Air Rhino, as discussed in the section Takeoff and Landing, it does present problems when compared with the design requirement from the Gold Mission Request for Proposals of a 300 ft diameter ground circle with 50 ft object clearance. The addition of flaps helps Air Rhino considerably, but landing distance still exceeds this constraint.

An analysis of the landing distance for various wing areas and plane weights appears below:

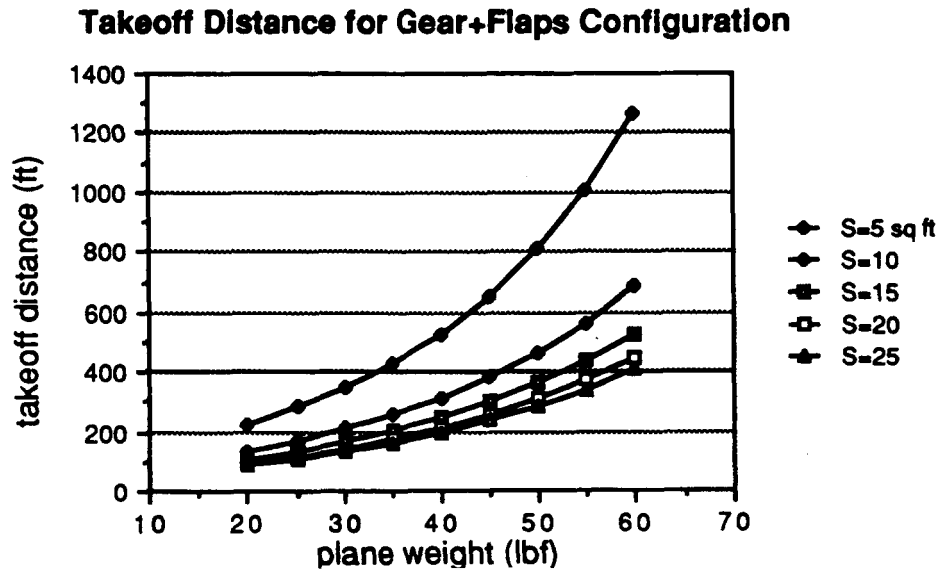


These charts are based on standard equations for estimation of approach distance, flare out, and ground roll.¹ A ground friction coefficient of 0.1 for soft grass, constant density, and an obstacle clearance of 50 feet are assumed. A flare out time of only 0.1 second is also assumed; this is adjusted for RPVs from the accepted value of 3.0 seconds. The 'gear + flaps' graph, using a flap system with $\delta C_{do}=0.06$, predicts a landing distance of 340 feet. This distance, however, is still greater than the proposal constraint of 300 feet (150 foot radius). As a contrast, the Air Rhino without the flaps is predicted to have a landing distance of 710 feet. Note that these

¹Leland M. Nicolai, Fundamentals of Aircraft Design, METS, Inc., Xenia OH, Chapter 10.

distances are based on a turf runway. If the runway surface is changed to something smoother, say asphalt, the landing distance will increase because the Air Rhino has no braking capability. The installation of brakes or a small drag chute on the RPV should be the subject of future study.

A graph of takeoff distance for various plane weights and wing areas is shown below. As with the landing equations, the critical parameters are weight and wing area.



The takeoff distance for the Air Rhino is less than its landing distance, only about 210 feet. Takeoff should not be a problem for the Air Rhino, thanks to its engine being equipped to fly at much higher speeds than those encountered at takeoff.

Flight Performance

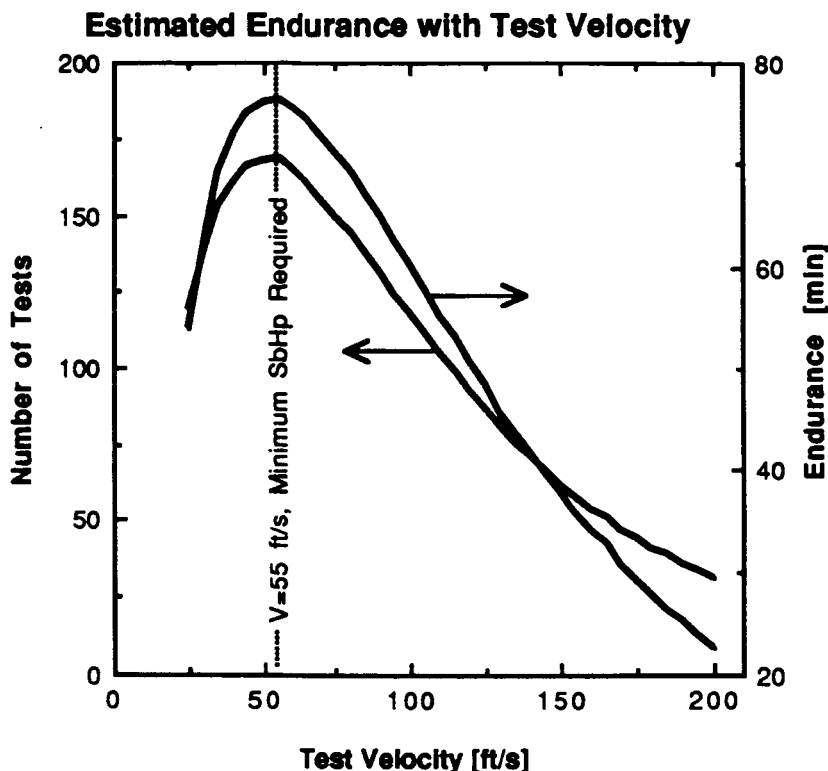
Air Rhino was not designed for high altitude flight, but high numbers for ceiling and rate of climb resulted from other ambitious performance goals, most importantly a V_{\max} of 200 ft/s in the dirty configuration. The immense power requirements for this flight regime make the Air Rhino extremely overpowered at cruising speeds, where excess power available can reach values of 4 Hp. The results are high ceilings and long range and endurance.

Range and Endurance

Because Air Rhino is not intended for a local flight plan, and not for cross-country flight, its distance range is relatively insignificant item in the list of performance results. However it is estimated that, when required, Air Rhino can fly up to 100 miles in a clean configuration (test airfoil $\alpha=0^\circ$) on a single 39 ounce tank of gasoline, at an airspeed of 80 ft/sec, or 55 mph. Even in a dirty configuration, with the test airfoil at a 30° deflection, the aircraft can travel 70 miles at 75 ft/sec (51 mph).

Air Rhino's endurance estimates are classified into two regimes--nonstop, cross-country travel at cruising airspeed, and local flights for the purpose of conducting multiple tests. As stated above, Air Rhino is not designed for cross country flight, but in clean configuration it can stay in the air up to 2:09 hours at an airspeed of 64 ft/s. In dirty configuration, the aircraft can fly for 1:42 hours at 55 ft/s.

A more meaningful indicator of Air Rhino's endurance capabilities is the duration and number of force balance test runs that can be performed on a single tank of gasoline. An estimate of these values as a function of test velocity is shown below for the aircraft in dirty configuration, with the test airfoil at 30° deflection:



These values were calculated assuming a total test duration of 7 s and a return time of 20 s. The return airspeed was assumed to be 55 ft/s, for minimum engine power required. After test runs at airspeeds above 55 ft/s, it was assumed that the aircraft would climb in order to exchange velocity for altitude, until reaching the return airspeed. Similarly, the aircraft would dive to gain speed between the return leg and the next test leg. These climbing and descending maneuvers were assumed to require only enough engine power to overcome drag at the mean airspeed of the maneuver, $V_{\text{test}} - V_{\text{return}}$.

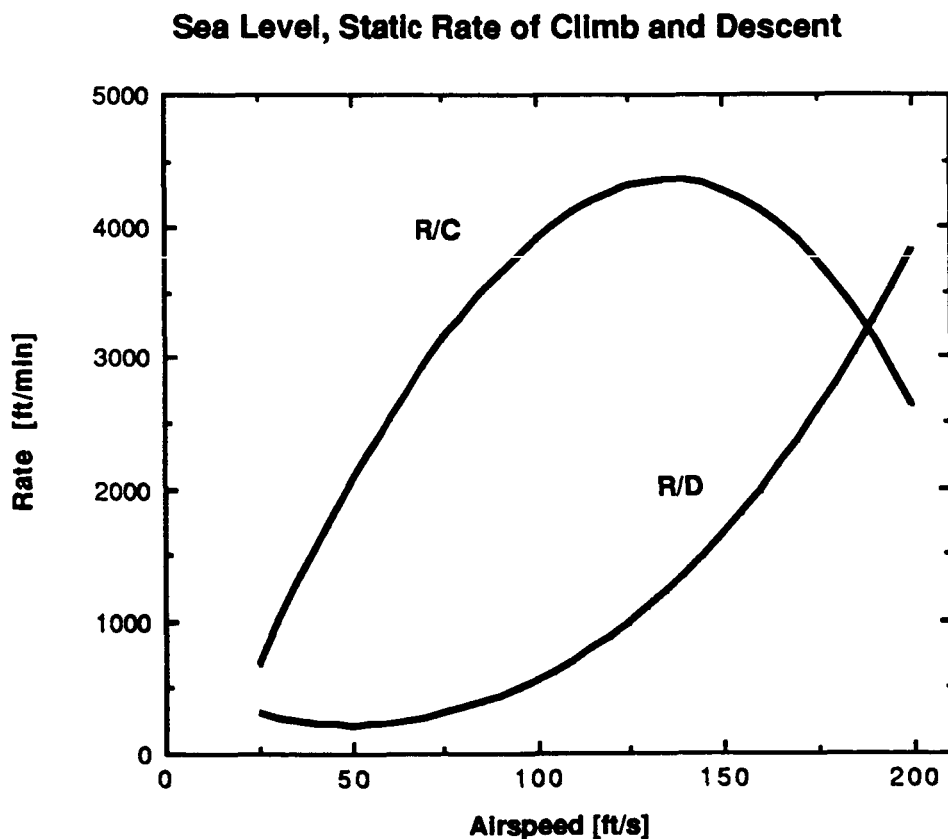
Air Rhino's propulsion system is designed for a minimum test session endurance of 30 minutes. Because fuel consumption is a function of total drag through its related engine power output, the fuel requirements for a 30 minute flight are dependent on the angle of attack of the test airfoil, and the velocity at which each test will be run. Air Rhino was designed for an endurance of 30 minutes at maximum test velocity of 200 feet per second, with the test airfoil at maximum angle of attack. Because its Q-82 engine

has a low fuel consumption rate of 1.3 lb/Hp/hr gasoline, Air Rhino can achieve this goal using only 39 ounces (1.82 lb) of gas. At 55 ft/s, which is its airspeed for maximum fuel efficiency, Air Rhino run tests for up to 101 minutes, more than 1.6 hours, with the test airfoil at maximum deflection.

For test flights at the opposite extreme, in a clean configuration, Air Rhino can make 50 test runs over a span of 34 minutes at V_{\max} . Maximum endurance increases to 240 tests in 95 minutes at a test speed of 64 ft/s.

Climb and Glide Performance

Air Rhino's sea level climb and glide performance as a function of airspeed in a clean configuration, with the test airfoil at 0° angle of attack, are illustrated below:



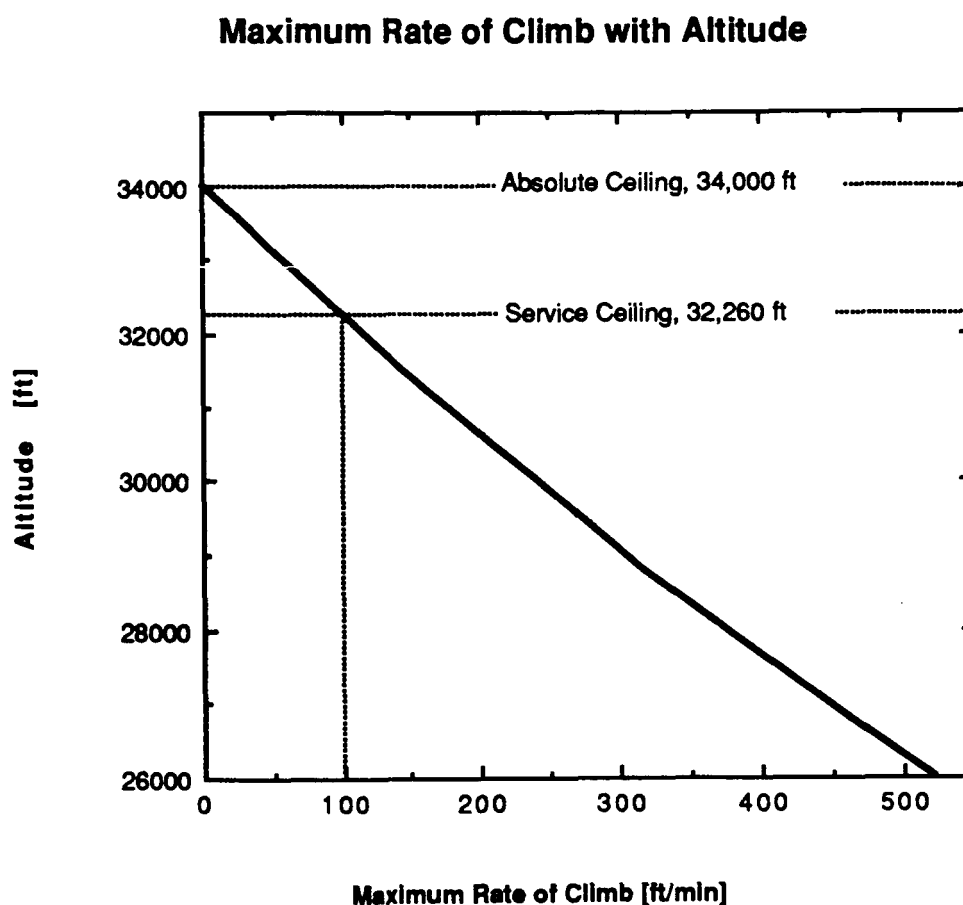
The graph shows a maximum rate of climb of 4,380 ft/min at an airspeed of

133 ft/s. The rate of descent increases with airspeed to a value of 3,840 ft/min at V_{\max} . Air Rhino's glide angle as a function of airspeed can be derived from this graph. The glide angle peaks with a value of 16 at $V = 55$ ft/s.

Not shown are climb and glide performance charts for a dirty configuration, with the test airfoil at 30° deflection. In this case, the maximum rate of climb decreases to 3,600 ft/min at an airspeed of 120 ft/s. The rate of descent becomes 6,300 ft/min at V_{\max} , and the maximum glide angle reduces to 13 at $V = 55$ ft/s.

Flight Ceiling

Whether in a clean or dirty configuration, Air Rhino's flight ceilings are very high, as shown below:



The above is an estimate of the aircraft's high altitude performance

based on power comparison, showing a service ceiling of 32,260 ft and an absolute ceiling of 34,000 ft. The drag increase for the dirty configuration lowers these values by only 7% (29,700 ft) and 10% (30,600 ft) respectively.

These altitudes are well beyond the one mile range of the telemetry unit currently expected for Air Rhino, so it is not suggested that the aircraft be flown at these altitudes without first being equipped with longer range telemetry. Further, manual controls are not practical at altitudes above 6,000 ft, so a preprogrammed flight computer would be required. Lastly, FAA regulations impose a flight ceiling on all RPVs, and special clearance is required to fly at altitudes above the legal limit.

Part IV. Operation

IV. Operation of the Proposed Plane

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System Operation

The Air Rhino has been designed to perform its mission with a minimum of work for the operators. Obviously there will be some changes as the Air Rhino design is further refined. But the knowledge of the configuration of the airplane and the experience gained during construction of the technology demonstrator allow a reasonable estimate of operation procedures.

The following section is a preliminary "operator's manual" for the Air Rhino, including maintenance and assembly instructions.

I. Transport

During transportation of the Air Rhino, protection of its fragile telemetry and data acquisition equipment is essential. Be sure that the locking screws on the force balance have been locked in place and do not transport the fuselage component with a test airfoil attached. Transport the fuselage component in its original foam shipping carton if possible.

The wing components are less fragile but care should be taken not to puncture their monokote skin surfaces. If any skin surface is accidentally punctured, use the provided "Surface Patch Kit" (q.v.).

The ground telemetry unit should be transported carefully as any computer equipment would be moved. Be sure the battery pack is fully charged before operation -- overnight if possible.

II. Assembly

Set the fuselage component on a level surface with landing gear retracted. (Turn the fuselage power system on and retract the gear using the control panel -- do not move the gear by hand.) Attach the two wing components to the fuselage component, making sure they are seated snugly on the retention spikes and the control cables are linked to the fuselage linkages. **BE SURE TO TIGHTEN THE LOCKING SCREWS!**

Check the engine to make sure its ignition and throttle lines are plugged in and tight. Check the onboard battery meter.

Make sure the control panel is within reach and the fuselage system is on. To

test the fit of the wings, lift the plane by its nose about 10 inches off the ground onto its tail section. The wings should move with little vibration. While holding the plane in this position, extend the gear using the control panel. Then gently lower the plane onto its rear wheels, and let the nose settle onto its front gear.

Calibrate the force balance (without a test airfoil attached) by following the instructions that came with the telemetry unit. This procedure will also check the telemetry and radio links. When finished, REATTACH THE LOCKING SCREWS and then slide the test airfoil onto the connecting spike. Be sure that the airfoil is near the plane surface but not touching it. Once the airfoil is attached, remove the locking screws and gently tap the airfoil to be sure its movement is not restricted.

Test the movement of the control surfaces and the nose wheel from the control panel. Also check the fuel level and add more if necessary. Make sure the test airfoil is set at 0° angle of attack.

III. Flight

As a test of the motor and landing gear, taxi the plane for a short distance and turn it around. The landing/takeoff area should be clear and fairly smooth -- a football field or a large parking lot would be acceptable. Be sure that onlookers are kept well away from the runway area. The Air Rhino can take off and land on a 400' runway surrounded by 50' trees or obstacles, but the less obstructions nearby, the better. It can also handle wind speeds up to about 20 mph.

At this point, the Air Rhino should be piloted like a conventional RPV. The landing gear need not be retracted unless high-speed (over 100 f/s) tests are planned. Climb the plane up to about 500' altitude to allow plenty of line-of-sight operation room. The estimated range of the control and telemetry systems is about one mile, so this should not be a problem as long as you can see the plane clearly. Do not fly the Air Rhino out of line of sight unless it is outfitted with the optional Safari extended telemetry package.

Once aloft, fly the Air Rhino in an elongated oval to get a feel for the controls. Vary altitudes, speeds and turn angles. Once the plane and pilot are warmed up, set the test airfoil angle of attack to the desired angle. Set the autopilot speed to the desired speed. When the autopilot is engaged,

the plane will fly in steady level flight, and should accelerate to the set speed for a test run. Steer the plane around so it can make a long straight run parallel to and above the runway. Once the plane is headed in the right direction, engage the autopilot and release the controls. The telemetry system should kick in once the autopilot is engaged, and will signal once enough data has been taken. At that point, release the autopilot and bank the plane back around to the beginning of the test run. Tests should take about 5 or 6 seconds. The autopilot may be left on for the return to the start point if desired -- the controls will steer differently, though, and pilots may have a little trouble getting used to it.

If the plane takes too long to run a test, try to dive the plane as it comes around for the test run. This will help bring the plane up to test speed more quickly. Be sure the plane is at a high enough altitude before you try this, and **BE FOREWARNED** that Air Rhino is exceedingly nimble at lower flight speeds.

The plane should be able to fly for at least 30 minutes- more if test runs are kept below 100 ft/sec. Be aware that there is no fuel warning and that the Air Rhino has very limited glide ability. Check the technical handbook for estimates of flight time as a function of average speed. After the expected flight time, or 90 minutes in any case, fly the plane around the oval path and make sure the test airfoil angle of attack has been reset to zero, the autopilot is engaged, and the landing gear has been lowered. Then bring the plane around and land. Because the gear does not have brakes installed, allow plenty of room for ground roll. The autopilot will keep the plane stable through wind gusts and ground effects. Again, make sure spectators are well back from the landing area.

IV. Maintenance

The Air Rhino will operate on standard model airplane fuel. The fuel tank access is under the rear top hatch on the fuselage. Be sure the ignition system is unplugged before refueling.

The control linkages may need an occasional spray of WD-40 or some other spray lubricant. The rear control surface linkages can be reached while the gear is down through the strut retraction holes on the wing undersides. The elevator linkages are visible when the wing components are removed. The servos moving the control surfaces are all mounted

within the rear fuselage and can be reached through the rear top hatch. DO NOT lubricate the force balance setup- it is a precision component and easily damaged.

Landing gear tires should be checked after each day of flights for flaws and flexibility. Replace if necessary.

The onboard battery plug is located on the right side of the plane near the nose. One charge should be sufficient for a day's testing.

The telemetry components contain no user-servicable parts and should not be tampered with.

V. Data Acquisition

The data obtained during test flights should be automatically saved after each run is completed. The software is fairly self-explanatory and is non-locked for your use. Air Rhino Systems recommends that you save at least one unmodified backup copy for use in case of accidents.

We hope you like the Air Rhino. Drop us a line and tell us what you think.

- The Design Team

The above manual is meant to be only a general guide, mostly because there are many unforeseen problems than can and probably will crop up between the design phase and the actual first test flight. One area where system operation has not been well defined yet is the type of computer and associated software needed to store and manipulate the data provided by the Air Rhino. However, since many colleges and corporate facilities (including Notre Dame) have developed similar software and hardware for use in gathering wind tunnel data, this should not be a problem. In fact, packaging the system hardware together with the Air Rhino may prove to be a profitable marketing ploy.

Another area needing further study is the 'piloting warnings' area. Until a full-scale prototype is built, and tested by experienced RPV pilots, only guesses can be made about the subjective handling characteristics that make the difference between an easy-to-fly hot seller and a turkey. While the technology demonstrator (q.v.) has yielded some data in this regard, obviously more evaluation will be necessary.

Maintenance Requirements

Expected maintenance

The maintenance requirements of the Air Rhino should be fairly similar to those experienced with other RPVs. Much of the technology is, after all, expected to be off-the-shelf. A breakdown of the expected requirements for each subsystem of the plane is given below:

DATA COLLECTION: The data collection system consists of the force balance, other sensors, and the telemetry package. The force balance, as with conventional balances, will require some calibration. It has not been worked out yet exactly how this calibration will take place. The other sensors will most likely be fairly durable, since most were built specifically to withstand the aircraft environment encountered by the Air Rhino. The telemetry package, as long as it is placed to avoid excessive vibration, should not need maintenance. This is good since the telemetry package is relatively difficult to access. The battery pack supplying the data and control systems will need to be recharged, but a large enough battery should supply all systems for a day's tests. A side-mounted plug socket will not add significant drag and will ease the recharging operation.

AERODYNAMIC SURFACES: The wings, stabilizers, and control surfaces should not need any maintenance other than occasional lubrication of the control linkages.

PROPULSION: The fuel tank will be supplied via a tank inlet pipe running from the rear top fuselage hatch to the fuel tank. Presumably, lubrication of the engine, either with a fuel-oil mix or separate oil introduction, will also be needed. The propeller is protected by the tail booms and landing gear, so replacement propellers should not be needed frequently.

STRUCTURE: The Air Rhino is made of high-strength, low-fatigue structural elements and thus should not need any maintenance.

LANDING GEAR: The retraction mechanism may need occasional lubrication. Gear tires will also need to be replaced at regular intervals, but the interval length will depend on type of tires and landing surface used.

Turnaround time

For each flight of the Air Rhino, several maintenance operations will need to be performed. Time estimates for these operations are listed below:

REQUIRED MAINTENANCE PER FLIGHT	
Refueling	3 minutes
Force Balance Calibration	5 minutes
Control Surface Check	1 minute
Change of Test Airfoil (includes locking and unlocking force balance)	5 minutes
TOTAL TIME	14 minutes

These times are probably fairly accurate with the exception of the force balance calibration, which is simply a best guess at this time. The turnaround time will be only 9 minutes if the same test airfoil will be used for the next flight. The force balance will most likely have to be calibrated each time, however, since the vibration and shocks of the RPV environment will have some effect on the balance from flight to flight.

Additional turnaround time between testing days must be allotted for battery recharging in both the Air Rhino and, if necessary, the power supply for the ground station.

System Safety

The concern for safety associated with the operation of a remotely piloted vehicle is of primary importance, as it is with the operation of any type of vehicle. To be considered truly safe, it is necessary that neither the Air Rhino nor anything else be damaged as a result of its operation. In a disaster situation, it should always be remembered that the safety of the Air Rhino should be of least concern in relation to its environment. Three areas of concern must be addressed when analyzing the safety of operating a remotely piloted vehicle. These three areas are positive control, unplanned descent, and collision avoidance.¹

Positive control of an RPV can be enhanced by a careful consideration of the vehicle's essential parts which provide guidance and control. A fair amount of redundancy should be applied to these parts of the vehicle which are critical to its safe operation. In this regard, the assurance of reliable control is more important to the overall concern for safety than is the vehicle's structural integrity. For positive control, it is necessary that the command signals from the ground operator reach the RPV. A primary problem which has resulted in bad command links is electromagnetic interference. A transmitting frequency well removed from the transmitting frequencies of public radio stations is recommended. Provisions should also be made in the design stage to allow for reestablishing a link that may become temporarily interrupted.

An unplanned descent is obviously something that is not desired. If an unexpected descent should occur because of some system failure, it is essential that the landing point can be controlled. The descent should be controlled so as to land in the least populated location. It would be best if the RPV were flown in an area devoid of people. The descent should be slowed down as much as possible to minimize the damage to objects on the ground caused by impact. For safety reasons, the descent should be slow, but at the same time the descent path should be steep to minimize the potentially-damaged region on the ground. A satisfactory compromise will need to be found to keep a sufficiently large descent slope, but at the same

¹Much of this material from RPVs -- Aerodynamics and Related Topics, Rhode Saint Ghenese, Belgium: Von Karman Institute for Fluid Dynamics, 1977.

time keeping a sufficiently slow descent to minimize possible damage to the environment. The autopilot system can be programmed to ensure proper guidance if an unexpected descent were to occur.

Collision avoidance is often considered the biggest technical challenge when it comes to safety. In civil air space, flight operations are conducted by a see-and-be-seen philosophy. There are several areas that fall under the area of collision avoidance: visibility, exact knowledge of the RPV location, air traffic control (ATC), and assigned air space operation. The use of lights and paint can increase visibility, but it is more difficult to make an RPV see another aircraft. If an RPV is flown at low altitudes away from an airport, which is the plan for this research vehicle, collision with an airplane should not be a problem. It is safest to assign restricted airspace to RPVs that other aircraft would not enter. RPV safety can be enhanced by only flying one RPV at a time. Especially in the case of an aircraft without transponders, i.e., a non-cooperating aircraft, a way to locate of the aircraft is important. Two possible options are imaging sensors, such as a TV, and active radar. Air traffic control is quite important for RPVs which will be operating in the same flight envelope as general-aviation aircraft, but for a research-intended RPV operated by line of sight, air traffic control is not a primary concern. Overall, the safest environment for RPV operation is one with the lowest possible population and amount of ground property which can be damaged.

Reliability for the overall safety of the RPV can be increased by the use of multiple engines. The tradeoffs associated with multiple engines, such as the addition of weight, would need to be considered depending upon the size and configuration of the RPV. Reliability would also be enhanced by the installment of a backup power supply. This power supply could help to keep various systems operational, such as the autopilot, in the event that the original power supply fails. Various emergency systems could also be constructed or added so as to decrease the extent of the damage. Such systems include parachutes, stowed rotors, pitched wings, and Magnus Effect wings. For this RPV a parachute was the only emergency system that was considered, but the parachute size was found to be too large to be feasible. Damage to the RPV and any object it may hit could also be reduced by the the use of touchdown load attenuators, such as airbags, which would minimize the shock loads encountered. Touchdown load attentuators were

not considered for the mission at hand.

Finally, regulations are imposed in the operation of most craft to ensure safety. Regulations exist for the operation of normal hobby-type RPV's, and these rules should be adhered to when operating this research vehicle. An RPV must be licensed in order to be operated. Often this licensing procedure is one of the most time-consuming steps in the first stages of operation. This licensing requirement must be acknowledged, and it is advised to begin this process as soon as it is permitted to do so.

Social and Environmental Impact

The Air Rhino will, if the design proves to be as efficient and useful as intended, be of great help to those interested in low Reynolds number testing. The Air Rhino, and other proposals to meet the Gold Mission requirements, are pioneering a new use for remotely piloted vehicles. As professional engineers, however, the design team has a responsibility to design products that are beneficial to society as a whole and not simply the products' buyers.

The Air Rhino will affect society and the environment in the following ways:

KNOWLEDGE: The Air Rhino will provide data on airfoil sections at low Reynolds numbers in realistic test environments, data that was before either very expensive or simply not possible to achieve.

SAFETY: While every effort has been made to minimize risk to those using the Air Rhino and innocent bystanders, there will always be some small risk with an RPV of damage after catastrophic failure. The small size and relatively light weight of the Air Rhino should minimize damage in case of such a failure. The pusher propeller configuration has the added benefit of protecting the propeller somewhat from wildlife strikes and stray fingers.

ENVIRONMENT: The Von Karman Institute for Fluid Dynamics states that "there are two environmental aspects within the use of RPVs ... the air pollution that is caused by engine emissions ... (and) aircraft noise." The Air Rhino uses about 2 pounds of fuel per half-hour flight, and a fair percentage of this fuel will be dissipated into the atmosphere. Ways to minimize this fuel loss must still be researched. Noise has been minimized through use of a multi-blade propeller and semi-internal engine mounting.

F.C.C. and OTHER REGULATIONS: Flight and licensing regulations for RPVs will have to be checked with the F.A.A. and the A.M.A., especially because of the experimental nature of the Air Rhino and the new use to which this RPV is being applied. F.C.C. regulations govern the choice and use of both control and telemetry systems, but these systems will most likely be, at least in part, bought from outside manufacturers who must comply with these same regulations.

AIR TRAFFIC: The Air Rhino flight plan is designed to remain well under the 1000 foot altitude level, so collision with larger planes should not be a problem. The relatively small market for this plane will mean only a slight influence on the number of RPVs in the air today.

Part V. Manufacturing and Sale

V. Manufacturing and Sale of the Proposed Plane

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Manufacturing Requirements

Production of the Air Rhino could be incorporated into a small business setup. A plan is presented here to specify the requirements of such a small business.

The design of the Air Rhino, because it simply utilizes a flightworthy craft to carry the test airfoil aloft, could be adapted fairly easily to a radio controlled (RC) recreational model craft. Yet the instrumentation additions and technical specifications for in-flight load data collection could still be incorporated in the sale of specifically outfitted RPVs.

Two groups working under one roof could effectively form "Air Rhino Rec. & Tech", a company organized to provide complete technically outfitted RPVs for various in-flight data collection purposes and model aircraft kits available for the RC flying enthusiast.

The RC group would be responsible for including materials and instructions for assembly of the Air Rhino model aircraft. Materials would include punch out balsa ribs, spar pieces, boom pieces and heavier sheeting pieces for reinforcement as needed. Mechanical cutting machinery would be used to stamp out the required airfoil shapes on a sheet of heavy balsa. Spar pieces could be cut with another machine. Boom materials could be either of spruce frame work or of high strength lightweight, composite tubing. The spruce needed for the spruce framework could be machined inhouse. The composite tubing could be bought from a manufacturer directly and simply included with the kit. Costs for the two possibilities are thought to be about the same. The kits would be distributed to hobby stores where RC flyers could purchase them. Staffing needs for this group would be the following:

- 1 Quality control engineer: inspects materials for defects, handles materials purchasing
- 2 General kit assembly workers: operate machinery, assemble kit sets and pack for shipping

The Technical group would be much more involved. Complete aircraft must meet the specifications of the customer. Machinery for this

group would include woodworking devices such as lathes, saws and sanders. Technical equipment would be needed to test the data acquisition system, including circuit boards for data reduction and conversion to computer software code. Specialists would be needed for the electrical hookups of the plane, including servos, receiver and data acquisition gear. The data collected would be tested and evaluated by this person as it is tied into the electronics. Propulsion and aerodynamic considerations could be handled by another specialist. These two people would also be responsible for modifying the basic design to meet specific criteria of the customer. Assisting in the assembling of the system would be 2 or 3 workers doing general tasks.

So the staff requirements here are:

- 1 Aerodynamics/Propulsion specialist: create custom applications
- 1 Electronics/Data Acquisition specialist: create custom applications
- 2-3 General staff assembly: operate machinery, build materials to specifications, assist in flight and ground testing

The two groups would share the following general staff:

- 1 Accountant: manages all financial affairs
- 1 Salesperson: handles advertising and sales
- 1 Office Manager: assists accountant and salesperson, handles shipping of products and incoming bills, some secretarial duties

The overall requirements for the company would be a staff of 10 or 11 full time employees sharing a large workroom facility, with at least five offices and some warehouse/storage capacity.

Productivity is assumed to be 3 Technical RPVs and 30 RC kits a month. Assuming an average monthly salary of \$2,600 and overhead at 60% labor cost, fixed costs would be \$45,760 / mo. Given the above output and the cost estimates for the Air Rhino in the Cost Estimates section, doubling materials costs for the custom Technical RPVs and subtracting data acquisition costs for the RC units, materials outlay works out to \$60,150 (RCs) + \$39,450 (RPVs) = \$99,600. So breakeven income is then \$145,360 per month. Pricing the RPVs at \$25,000 and the RCs at \$2800 yields cash

income of \$84,000 (RCs) + \$75,000 (RPVs) = \$159,000 a month.

These numbers are very rough estimates and could change once the market for each product is analyzed and tapped. However, this example illustrates the potential profitability of the Air Rhino.

Cost Estimate

The following is a break down of the cost estimate of the Air Rhino:

DATA ACQUISITION		
	absolute pressure transducer	\$900
	angular displacement transducer	\$320
	circuit board (on board)	\$1000
	circuit board (ground station)	\$1000
	differential pressure transducer	\$600
	force balance	\$100
	linear variable displacement transducer (2)	\$200
	strain gages (4)	\$200
	thermistor	\$200
	other (battery, mounts, etc.)	\$50
	data acquisition total	\$4570
PROPULSION		
	gas engine	\$400
	fuel tank	\$10
	mount	\$10
	other(fuel, etc)	\$25
	propulsion total	\$445
CONTROLS		
	automatic pilot	\$600
	receiver	\$100
	servos (11)	\$440
	transmitter	\$200
	controls total	\$1360
STRUCTURE		
	mylar skin	\$20
	tail booms	\$20
	wood	\$100
	structure total	\$200
	TOTAL	\$6575

Of note here is that the cost does not include overhead such as the manufacturing costs, labor costs, and costs associated with the work place

(rent, heat, upkeep, etc). These costs can be added in, but this cost estimate is purely based on the cost of the remotely piloted vehicle itself. So, it allows one to know exactly how much the physical plane could cost if it was actually built.

A guess as to overall costs of the plane, if labor is figured at \$25/hr for 200 man hours and overhead is figured at 60% of labor, would be \$14,575.

As can be seen by this table, the main contributor to the overall material cost of the Air Rhino is the data acquisition package. This does **not** include the cost of a personal computer that either must be provided by the purchaser, or can be packaged with the Air Rhino for additional cost to the consumer. The above cost estimate does include the necessary circuit board that is installed in the ground computer. Finally, this is only an estimate of the proposed cost. As the design is more finalized and specific brands of instruments are chosen, the cost could either go up or down accordingly.

Technology Demonstrator

The original proposal for the technology demonstrator will first be presented followed by the description of the final built up model. Much of what is presented here is the result of lengthy conversations over the telephone and discussion during the Toledo Show with Mr. Keith Shaw, an employee in the biophysics department of the University of Michigan. He is a "guru" in the realm of electric radio controlled model aircraft.

Note that in the technology demonstrator writeup, the "demonstrator" refers to the small model; the "proposal" refers to the full-scale actual product.

Design

The demonstrator is ideally supposed to be a scale replica of the actual RPV proposal design, to test accurately the viability of the design concept. Major problems were encountered, however, due to the huge scale difference between engines in the proposal and the demonstrator. The proposal engine specifications call for 8 Hp while the Astro 15 used in the demonstrator yields at most 1 Hp. A 1/8 scale plane, however, would be impossible to build due to size and weight requirements imposed on the fuselage by the Astro 15's battery pack. Given this problem, a directly scaled demonstrator was scrapped in favor of a plane with the same general configuration. This plane would at least validate our design as tenable if it made it off the ground.

The total weight of the technology demonstrator was determined by the weight of the power system, that is, the weight of the motor and battery combination. To yield glow-engine like performance, the total weight of the aircraft should not be more than twice the weight of the motor and battery together.

The weight of the motor and battery was found to be 32.0 oz. Using a weight percentage approach, this meant that the total weight of the aircraft would be 64 oz. or 4 lbs. To find the weight of the airframe alone, hinged and covered with no electronics, battery or motor; the weight of the motor, battery, two servos, receiver, receiver battery pack and speed controller was deducted from the total weight of the model. This yielded the hinged and

covered airframe weight of 20 oz. or 1.3 lbs.

This 1.3 lb airframe weight is approximately 32% of total model weight. The key to achieving such a low airframe weight is a built-up structure (with either ladder or Warren type bracing) and some sheet balsa skinning for added strength.

The airframe weight can be nudged to a higher value, while still satisfying the 50/50 weight distribution between propulsion system and the airframe and controls systems, and by using micro-servos. The micro-servos weigh less than half of the supplied standard servos, saving a total of 1.7 oz. This would allow a further 1.7 ozs increase to the airframe weight to 21.7 oz.

The desired wing loading for the aircraft would lie between 15-20 oz/sq ft, as the majority of radio controlled sport aircraft have wing loadings within this range. Such aircraft have good flying characteristics. Note, that although the wing loading for the proposal is over 32 oz/sq ft, the demonstrator cannot hope to meet that with the Astro 15 system. To reduce the planned weight of the structure, the upper limit of the wing loading was used. Such a wing loading, together with the total weight of 4 lbs, yielded a wing area of 3.2 sq ft or 460. sq ins.

Typical values of wing aspect ratios for RPVs like the demonstrator ranged between 5.5 to 6, just under the proposal aspect ratio of 6.33. An aspect ratio of 5.5 yields a wing span of 50.30 ins and a chord length of 9.15 ins. An aspect ratio of 6 yields a wing span of 52.60 ins and a chord length of 8.75 ins. An off-the-shelf wing kit is available from Great Planes Model Mfg. Inc. that has dimensions very similar to that required by the demonstrator. The wing kit has a wing span of 52.0 ins and a chord length of 9.8 ins, which yields a wing area of 510 sq. ins. This is slightly more wing area than what is required.

There were three options at this point:

- Use the stock wing without any modifications and enjoy the benefits of a lower wing loading of 18 oz/sq in. This would have to depend on whether the roughly 50/50 weight percentage split between propulsion and everything else could be maintained.

- Reduce the wing span of the model so that we may achieve the upper limit of the wing loading range. This would yield a modified wing span of 47.0 ins. The resultant aspect ratio would only be 4.8. This would

require further research into the effects of such a low aspect ratio on flight characteristics of a model this size.

- The third option lies between the two above. There are 5.0 ins. of free play for the wing span or approximately 49 sq ins. of wing area while maintaining the same chord length. It was decided not to fiddle with the chord length since this would alter the profile of the airfoil.

Typical RC values for total horizontal stabilizer area ranges from 15%-20% of wing area, with the design proposal coming in at 22% Vertical tail area values are typically 10% of wing area, with the proposal at 6% Since the wing area is not yet finalized, the area of the empennage cannot yet be accurately determined. Assume a wing area of 460 sq ins. This would yield a minimum horizontal tail area of 70.0 sq. ins. and vertical tail area of 46 sq. ins. This 46 sq. ins. is divided by two to obtain the vertical tail area for each of the twin booms. Typical values of vertical and horizontal tail volume ratios need to be confirmed before tail boom lengths can be determined.

Ballpark figures for power required for steady level flight of RPVs can be found by the following formula:

$$\text{Power Req. (Watts)} = \text{Total Model Weight (pounds)} \times \text{Wingloading (oz/sq.ft.)}$$

The units on the L.H.S. do not agree with the R.H.S. of the equation. This means the above equation probably determines a ball park figure for take off power required. Thus, for the figures determined so far for the technical demonstrator, the power required for steady level flight for a) 18 oz/sq ft configuration is 72 Watts and b) 20 oz/sq ft configuration is 80 Watts. As a general rule of thumb, the power required for takeoff is twice that required for steady level flight. This is assuming a paved, blacktop runway. Grass field type runway would require more power for takeoff.

Given the motor battery pack is rated at 16.0 volts with 1.2 ah. capacity. This would imply that we would need to draw more than 160 watts for takeoff from the battery pack to compensate for the various component inefficiencies in the powertrain of the electric motor. Electrical power being a product of voltage and current determines the current draw required to achieve that power output. The current draw is determined by

the size and pitch of the propeller used. The larger the propeller, the larger the current draw. Thus, we would need a propeller that draws a current of more than 10 amps static (current drawn by bench mounted motor turning a particular prop). The above assumes a paved runway. A grass runway may require 1.5 to 2.0 times as much power hence 1.5 to 2.0 times the current required for the paved runway takeoff.

The endurance of the model would depend very much on the power setting at which it is flown at. If the model is flown with a current draw of 20 amps, the endurance would be approximately 3.6 minutes at best. The lower the throttle setting, the greater the endurance.

Because of the importance of a low weight structure for the successful flight of this model, much, if not all of the model was built up of thinly and balsa wood. Lightening holes were drilled wherever possible to reduce the weight of the airframe.

Although the tail length had been decided through stability requirements, the total length of the fuselage has yet to be determined. The length of the fuselage forward of the C.G. will be decided once the weight of the empennage and tailbooms have been determined after construction. This way we will know approximately how far forward the 1.5 lb battery needs to be to balance the weight located behind the C.G.

In the interest of saving weight. Only two servos will be used. One will control the elevator and the other will control the aileron. Yes, that is correct, only one aileron will be used for directional control. Servo extension chords need to be procured for this servo mounting configuration. The above set up not only saves weight in terms of using fewer control surface linkages and hinges, but it will also be most favorable for weight distribution in this twin boom design. We can locate the elevator servo in one boom with some form of control linkage, that has yet to be decided, running to the rear. Then the aileron servo can then be located in the other boom to actuate the aileron using a push rod, torque rod configuration. Hopefully, this will give us a close to symmetrical weight distribution along the axis of the fuselage pod. The draw back in this set up is that it does not provide directional control while on the ground unless a separate servo is used for steering the nose gear.

Summary of Weights and Dimensions of the Proposed Technology Demonstrator

	---estimated---	---actual---
Total Weight	4 lbs	5 lbs
<u>WINGS</u>		
Wing Area	460 sq.in.	510 sq.in.
Wing Loading	20 oz/sq.ft.	22.58 oz/sq.ft.
Wing Span	47.0 in.	52.0 in
Wing Chord	9.80 in.	9.8 in.
Wing Planform	Constant	Constant
Wing Aspect Ratio	4.8	5.33
Wing Location	High	Low
Dihedral (each wing tip)	1/2 in.	2.0 in.
<u>HORIZONTAL STABILIZER & ELEVATOR</u>		
S_s/S_w %	22.0%	23 %
Horizontal Stabilizer Area	101.0 sq.in.	117.81 sq.in.
Horizontal Stabilizer Chord	7.0 in.	7.7 in.
Horizontal Stabilizer Span	14.5 in.	15.3 in.
Horizontal Stabilizer Aspect Ratio	2.1	1.987
S_e/S_s %	20.0%	25.9 %
Elevator Chord	1.4 in.	2.0 in.
Elevator Span	14.5 in.	15.3 in.
Tail Length (C.G. to tail A.C.)	22.3 in.	26 in.
Horizontal Tail Vol. Ratio	0.5	0.6128
<u>VERTICAL STABILIZER</u>		
S_v/S_w %	10.0%	11.2 %
Total Vertical Tail Area	46.0 sq.in.	57.24 sq.in.
Vertical Tail Vol. Ratio	0.21	28.62 sq.in.
Area of each Vertical Stab.	23.0 sq.in.	5.4 in.
Vertical Stabilizer Ave. Width	5.6 in.	5.3 in.
Vertical Stabilizer Height	4.10 in.	0.29
Rudder	N.A.	N.A.
<u>HARDWARE</u>		
Motor Size	Astro 15 Cobalt	
Battery Pack	16 V, 1.2 ah	
Landing Gear	Tricycle	
Wheel Material	Plastic	
Tire Material	Foam	
Nose Gear Tire Diameter	1.5 in. diameter	
Main Landing Gear Tires	2.25 in. diameter	
Landing Gear	1/8 in. piano wire	
Control Functions	Elev, Aileron, Motor	
Aileron Actuation	Torque Rod	
Elevator Actuation	Pull-pull Cable	
Motor Actuation	Speed Controller	
<u>SOFTWARE LIST</u>		

Fuselage Pod	3/32 Bal, 1/64 Ply Doubler 1/2 Triangular-Stock 1/8 Bal, Soft Bal Block 1/16 Ply Bulkhead 1/6 Bal Top Sheet
Tail Boom	1/8 Bal. Brace 3/32 Bal. Lam 1/6 Ply. Lam.
Landing Gear Blocks	1/8 Bal. Rib, 1/32 Bal. Lam
Horizontal Stabilizer	1/8 Bal. 1/32 Bal. Lam
Elevator	1/8 Bal. Sheet
Vertical Stabilizer	

Special Considerations

The final executed technology demonstrator at roll out did not differ drastically from its original conception. The pusher configuration with twin tail booms was maintained as it appears in the design proposal, though many ratios remained changed from the proposal. It was felt that it was best to utilize the maximum wing area (510 sq.in.) as provided in the PT-20 wing kit. Only two control surfaces utilized were elevator for pitch control and aileron for roll control. The two controls when used together can provide the RPV with turn performance. Throttle control was also provided through means of an electronic speed controller. Rudder control was not included in an attempt to keep the overall weight of the aircraft to a minimum. Tricycle undercarriage gear was utilized on the technology demonstrator as planned in the proposal.

Some changes were made during the course of construction when we realized that there were more labor saving R/C products available on the market that were applicable to the building of the technology demonstrator. The first change that was made to the technology demonstrator was the use of fiberglass tailbooms instead of fabricating them out of wood. These fiberglass tailbooms were intended for use on R/C helicopters. The full length (31.2 in.) of each boom weighed only 3.2 ozs. and were extremely strong and rigid for their weight. These were mounted into the wing using balsa block and plywood sandwich type construction.

The main spars supplied with the wing kit were made of basswood. This proved to make the wings far too heavy to achieve the 20 oz. structural weight desired. We substituted the basswood spars that was supplied with the kit for balsa wood spars. We used shear webbing to compensate for the use of the softer wood.

It was the original intention to use the full length of the tailbooms to provide for a favorable tail volume ratio. However, given the length of the fuselage, the motor battery together with the receiver battery placed as far in the nose as possible, would not balance the aircraft on the desired center of gravity location on the wing (approximately $0.3C_w$). It was at this time Joe Mergan, resident RPV expert at Notre Dame, suggested making the tail length two and a half times the wing chord length. This reduced the tail boom length from the full 31.5 ins. to about 24.0 ins. and permitted the center of gravity to be located on the main spar of the wing and not behind it.

An interesting feature of the technology demonstrator was the use of iron-on hinges for all control surfaces. This provided for a sealed hinge line which did not allow for any leakage of airflow between the upper and lower wing surfaces. The advantage in such a configuration is that it provides for a more efficient airfoil and greater control effectiveness due to little, if any leakage.

Probably the largest deviation from the original technology demonstrator proposal was the final ready to fly weight. The goal was a ready to fly weight of 4 lbs. The final constructed model was 5 lbs even though many weight saving building techniques were applied. This was 25% greater than the original intended weight. This led to the somewhat high 22.58 oz per sq.ft. wing loading. This was unfortunate since it ultimately degraded the take off performance of the aircraft.

Flight Test Results

The flight test plan was to first fly the demonstrator without a test section to test takeoff and landing performance. Then the test section would be added, and the control and stability of the plane tested in subsequent flights. An alternate propeller was also going to be tested.

It was felt that the small runway size was a potential problem. The runway area was a small parking lot ringed by a steel cable fence. If the plane could get off the ground, it should fly well. Or so we thought.

Flight Test Results

The demonstrator was first taxied on the ground to give its pilot a feel for its ground roll behavior. After he had ascertained that it was good, he

lined up the aircraft with the 'runway' and applied full throttle. The aircraft rotated after approximately 30 feet and proceeded to climb. After a steady climb to about 7 feet the aircraft suddenly dipped and the pilot felt that the aircraft would not be able to clear the chain barrier at the far side of the car park. The pilot then chose to abort the flight, not realizing that the ground roll distance would not be adequate for the aircraft to come to a stop.

In any case, the pilot proceeded to cut the throttle. On touch down, the aircraft veered left and rolled for a distance of approximately 15 feet before wrapping itself around a metal barrier. The metal pole took out the left wing between the wing root and tail boom. Repairs are currently under way at time of writing. Although the aircraft probably can maintain sustained flight under electric power, the aircraft is currently undergoing conversion to glow engine propulsion.

First, it must be admitted that the flying sight was somewhat less than ideal. Should the abort distance for takeoff have been decided before a takeoff attempted? Could the aircraft been nursed over the chain barrier and landed on the grass? Why did the aircraft veer left on touch down? Was an attempt made by the pilot to steer right, where there was so much room, to avoid the countless metal pole barriers on the left? Why did the aircraft not perform as well as it did the night before during taxing tests? All these questions remain unanswered since they are all a product of 20-20 hind sight. Hopefully, future design teams will address such possibilities before they occur so that proper contingency plans may be made.

As far as meaningful results, several pieces of knowledge came out of the technology demonstrator, despite its apparent lack of success. The plane did in fact fly straight and level, suggesting that the plane was both statically and dynamically stable. Some means of locking the nose wheel in the front position would have probably saved the plane, and future design iterations should keep this in mind. The test also underscored the need for reasonable wing loading.

One overall lesson Design Group E learned the hard way was that "it ain't over till it's over." Unforeseen circumstances can invalidate whole phases of design. As can be seen from the list of proposed and actual parameters for the technology demonstrator, the manufacturing phase of design can wreak havoc on the most carefully designed -- on paper -- plane.

Marketing

Any new product in today's competitive marketplace must sell itself if it is going to be worth anything to its manufacturer. The design team feels that there are several features that will make the Air Rhino an easy concept to market.

First of all, it's cheap. To do the same testing in a wind tunnel is expensive and near-impossible to get accurate results. The wide Reynolds number test range of the Air Rhino makes it even more of a bargain.

The Air Rhino is easy to operate. It should not take any special piloting skill to fly in a reasonable environment, and the autopilot feature makes coordinating smooth, accurate test runs a breeze. Its long endurance allows several test runs in one flight to minimize maintenance and maximize data acquired for time spent. Assembly and maintenance procedures have been designed to be simple and as foolproof as possible. The vulnerability of the force balance is a problem, but emphasis on the direct measurement of data (and no cumbersome data reduction) should make this a selling point. Portability of the entire system also allows customers in densely populated areas the chance to run real tests of airfoils without finding or building a wind tunnel, or having to drive two hours to find a suitable test point.

Simple, durable, reliable. That's Air Rhino. A six-page 'glossy brochure' mockup is available to illustrate further what the design team sees as a marketable aircraft.